# NACA

# RESEARCH MEMORANDUM

A COMPARISON OF THE EXPERIMENTAL AND THEORETICAL LOADING OVER TRIANGULAR WINGS IN SIDESLIP

AT SUPERSONIC SPEEDS

By John W. Boyd

Ames Aeronautical Laboratory Moffett Field, Calif.

# NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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#### SUMMARY

The results of an experimental investigation of the pressure distribution over two triangular wings in sideslip at supersonic speeds are presented. The two wings had identical plan forms, 45° sweepback of the leading edge, and an aspect ratio of 4.0. One model was composed of round leading-edge sections (NACA 0006-63) and the other of sharpnose, symmetrical, circular-arc sections. For both wings the maximum thickness of streamwise sections was 6 percent and was located at the 30-percent chord. The experimental data were obtained through a sideslip range from 0° to 9°, for angles of attack from 0° to 10°, at Mach numbers from 1.20 to 1.70 and at a constant Reynolds number of 1.8 million.

At the high angles of attack, the results showed a significant departure of the variation in the experimental loading with sideslip from that predicted by the linearized theory. In the lower speed range (Mach lines swept ahead of the leading edge) the positive dihedral effect predicted by theoretical calculations was in evidence at the low angles of attack but was not realized at the high angles of attack. The airfoils exhibited a variation of dihedral effect with lift coefficient similar to the variation with Mach number predicted by theory, in this case a loss of positive dihedral effect.

In the higher speed range (Mach line swept behind the leading wing), because of a detached bow wave ahead of the leading edge and resulting subsonic velocities at the leading edge, the load distribution revealed characteristics similar to those experienced in the lower speed range wherein the Mach lines were swept ahead of the leading edge.

#### INTRODUCTION

Reference 1 has shown that transonic flow effects have a significant influence on the experimental load distribution over triangular wings at high lift coefficients at supersonic speeds. These transonic effects have been shown to be primarily a function of the Mach number component perpendicular to the swept leading edge (reference 1). It is expected, therefore, that these effects may significantly influence the load distribution over triangular wings in sideslip. The present investigation was undertaken to determine whether or not these expectations would be realized and to provide data for a comparison of the theoretical and experimental load distributions. It is suggested that the reader refer to reference 1 for a more detailed explanation of transonic effects since, in the interest of brevity, some of the explanations concerning transonic flow characteristics are omitted in the present report.

#### SYMBOLS

- $\frac{b}{2}$  semispan, feet
- c local wing chord, feet
- cr root chord, feet
- mean aerodynamic chord measured parallel to the plane of symmetry  $\left(\frac{\int_0^{b/2} c^2 \, \mathrm{d}y}{\int_0^{b/2} c \, \mathrm{d}y}\right)$ , feet
- M free-stream Mach number
- P pressure coefficient  $\left(\frac{p-p_0}{q_0}\right)$
- $\frac{\Delta p}{q_0 \alpha}$  loading coefficient per unit angle of attack  $\left(\frac{p_1 p_U}{q_0 \alpha}\right)$ , per degree
- p local pressure on airfoil, pounds per square foot
- po free-stream static pressure, pounds per square foot

 $q_0$  free-stream dynamic pressure  $\left(\frac{1}{2} \rho V^2\right)$ , pounds per square foot

- R Reynolds number based on mean aerodynamic chord
- V velocity of free stream, feet per second
- w half-width of plan form at any root-chord position, feet
- x,y rectangular coordinates in the plane of the wing with the origin at the apex of the wing and x measured parallel to plane of symmetry of the wing
- angle of attack of wing at plane of symmetry, degrees
- β sideslip angle (positive when sideslipping to right), degrees
- € vertex half-angle of wing plan form, degrees
- $\mu$  Mach angle ( $\sin^{-1}\frac{1}{M}$ ), degrees
- ρ mass density of free stream, slugs per cubic foot

#### Subscripts

- l conditions on lower surface of airfoil
- u conditions on upper surface of airfoil

#### APPARATUS AND MODELS

The experimental investigation was conducted in the Ames 6- by 6-foot supersonic wind tunnel which is of the closed-return variable-pressure type with a Mach number range from 1.15 to 2.0. This wind tunnel is described fully in reference 2.

A sketch of the 45° swept-back triangular wing models which gives all plan-form dimensions is shown in figure 1. In order to obtain as high a test Reynolds number as possible, the maximum size model which was free from wind-tunnel-wall interference at the lowest test Mach number was used.

Since in reference 3 there was shown a pronounced effect of chord-wise-airfoil-thickness distribution on the flow characteristics of airfoil sections at transonic speeds and, since in the present experiment it was expected that similar transonic effects would be manifest, two different airfoil sections were selected for the wings. One wing was composed of round-nose airfoil sections 6 percent thick in streamwise planes. The section used for this wing was the NACA 0006-63 profile. The other wing was composed of sharp-nose, symmetrical, circular-arc sections. For both wings the maximum thickness of streamwise sections was 6 percent and was located at the 30-percent chord. See table I for airfoil ordinates.

The models were cast of bismuth—tin alloy and coated with zinc chromate to give a smooth surface. The cone which joined the wing to the support sting (fig. 2) was designed to minimize the pressure disturbance and at the same time fulfill the strength requirements. The support sting itself served as a conduit for the pressure tubes.

The right wing panel was fitted with 86 pressure orifices, each 0.013 inch in diameter, arranged to measure the local pressures on the surface in such a manner as to permit the calculation of the pressure difference between the upper and lower surfaces. These orifices were located in planes perpendicular to the plane of symmetry at three chordwise stations (fig. 1). These stations hereafter designated as stations 1, 2, and 3 were located at 25, 50, and 75 percent of the root chord, respectively.

The models were mounted with the plane of the wing in the vertical plane in the test section. A cantilever sting support was used as shown in figure 2. The sting angle of attack could be adjusted to any angle between ±5° while the tunnel was operating. Through the use of bent stings, various angles of attack and angles of sideslip could be obtained. For the present test both a 5° and a 10° bent sting were employed to give an angle—of—attack range of approximately 0° to 10° and angles of sideslip from -9° to +9°. Since the model angle of attack during the test was influenced by the deflection of the model support under load, an arrangement of mirrors and lenses was used to determine optically the true angle of attack. The angle of sideslip was determined by means of a cathetometer.

#### METHODS

#### Theoretical

The theoretical loading per unit angle of attack was calculated using the formulas of reference 4 as derived from the linearized theory. The flow field of a flat lifting triangular wing in sideslip is of conical form; that is, quantities such as pressure and velocity are constant along rays emanating from the apex of the wing. The flow, therefore, when shown in transverse planes (planes normal to the axis of symmetry) has a characteristic of two-dimensional flow in that the pressure plots at all fore and aft locations will be similar.

Since the theory is based on linear differential equations, the principle of superposition applies so that the pressure distribution due to airfoil thickness has no influence on the pressure distribution due to angle of attack or vice versa.

### Experimental

Tests.— In the present investigation, data were obtained over a Mach number range from 1.20 to 1.70 at a constant Reynolds number of 1.8 million. Measurements were made through a sideslip range from 0° to 9° and at angles of attack from 0° to 10°. It should be noted here that the load distributions presented in figures 3 through 8 over the trailing wing panel were actually measured over the leading wing panel at equal negative sideslip angles.

Recording and reduction of data. The pressures were indicated on multiple—tube manometers which were photographed to record the pressures. The data were reduced directly to spanwise plots of the pressure coefficient through use of a plotting machine.

Precision.— Surveys of the tunnel air stream (reference 2) have shown that at Mach numbers other than 1.40 there exist pressure and stream angle disturbances in the air stream. These surveys indicate, however, that the flow in the air stream is two-dimensional; that is, there are no appreciable transverse pressure gradients in horizontal planes. In the present test, therefore, since there is no cross flow in horizontal planes in the test section, the model was mounted with the wing in the vertical plane to minimize the effects of stream irregularities on the load distribution. Hence, since the flow was similar in all vertical planes, the static-pressure corrections applied were those measured in the vertical plane at the center line of the tunnel.

In applying these corrections it was assumed that the static pressures on the upper and lower surfaces were equally affected by the stream static-pressure variation and that the lifting pressures were not affected. These assumptions have been shown to be valid by the results of the investigation of reference 2.

The major items which may cause inaccuracies in the experimental pressure distributions have been noted in reference 5. Since the techniques employed in this investigation parallel those used in reference 5, the over—all precision should be approximately of the same magnitude; that is, the wing static pressures should be accurate to within ±1 percent of the dynamic pressure.

As noted previously, the size of the wing was chosen so that at the lowest Mach number there was no interference between the wing and the compression or expansion waves originating on the model and reflecting from the tunnel walls.

Errors made in measuring the angle of attack were confined to purely mechanical inaccuracies since the variation of the stream angle in transverse planes was negligible. A possible error of  $\pm 0.05^{\circ}$  in the angle of attack was incurred in the initial referencing of the model with respect to the stream direction. The angle of attack under load, determined by means of the optical measuring system, could be read to within  $\pm 0.03^{\circ}$ , resulting in a total possible error of  $\pm 0.08^{\circ}$  in the angle-of-attack reading.

Errors made in measuring the angle of sideslip were a combination of stream—angle variation and mechanical inaccuracies. Because of the axial variations in the stream angle at Mach numbers other than 1.40 it was difficult to determine the effective model angle of sideslip. At a Mach number of 1.20 the angle of sideslip may be in error as much as 1.1° due to stream inclination, whereas at a Mach number of 1.70 the stream angle may be as great as 1.1° in the opposite direction. All of the data presented herein are for angles of sideslip which have not been corrected for stream inclination. Mechanically the inaccuracy of the sideslip angle was limited to the accuracy of the cathetometer which could be read to within ±0.05°.

The absolute humidity of the air in the wind tunnel was kept below 0.0003 pound of water per pound of air at all times so that it had negligible effect on the experimental results.

# RESULTS AND DISCUSSION

The experimental results in the form of pressure coefficients on the upper and lower surfaces for the complete range of test variables

are presented in tabular form for any analysis which the reader may wish to make (tables II-VII). A portion of the data of station 3 was omitted because of noticeable interference effects from the support cone. For the purpose of discussion in this report, figures showing the experimental load distributions are presented for Mach numbers of 1.20 and 1.70 and compared with the theoretical results. These data are considered representative of the results obtained throughout the test range.

### Experimental Load Distributions

In the analysis of the experimental lifting pressures the data show, as in reference 1, that, while theory and experiment did not always agree on the magnitude of the pressures on the wing, the experimental lifting pressures were essentially constant along rays from the apex of the wing so that conical flow fields existed for both theory and experiment. It is possible, therefore, and convenient in considering the loading over triangular wings to resort to transverse pressure plots since they are essentially similar at all fore—and—aft locations. Components of the velocity perpendicular to rays are considered in analyz—ing these transverse pressure plots. It may be seen that on the surface of wings moving at supersonic speeds the components may be either subsonic or supersonic, depending on the stream Mach number and the sweep of the ray considered, so that no inconsistency exists in referring to subsonic velocity components on the surface of an airfoil moving at supersonic speeds.

The variable  $\tan \varepsilon/\tan \mu$ , where  $\varepsilon$  is the semivertex angle of the wing and  $\mu$  is the Mach angle, is sometimes used to indicate the relative position of the Mach line from the apex with respect to the wing leading edges. Values of  $\tan \varepsilon/\tan \mu$  greater than one correspond to the condition wherein the Mach line is swept behind the leading edge and values less than one correspond to the condition wherein the Mach line is swept ahead of the leading edge.

Mach lines swept ahead of the leading edge.— The variation of the experimental load distribution with angle of sideslip for both the round—nose and the sharp—nose airfoil is presented in figure 3 for low angles of attack (approximately 2.5°) and for a Mach number of 1.20. The data presented here are typical of the results obtained at low angles of attack in this speed range (Mach lines swept ahead of the leading edge). Examination of the data revealed that, as the angle of sideslip was increased from 0° to 9°, the experimental data showed the increase in loading over the leading wing panel—and decrease in loading over the trailing wing panel expected from theoretical considerations, that is, a positive dihedral effect. Positive dihedral effect is

considered to exist when the loading is such as to cause the leading wing panel to rise and the trailing wing panel to lower.

For higher angles of attack (approx. 8.60), very little variation in the loading with angle of sideslip was noted. As shown in figure 4. the data obtained for both the round-nose airfoil and the sharp-nose airfoil indicated little change in the magnitude or the distribution of the experimental loading coefficients as the angle of sideslip was increased from 0° to 9°, very little dihedral effect being present in this case. Theoretical studies, while strictly applicable to only small angles of attack, lead one to expect that positive dihedral effect also would be present here. The reasons for the departure of the variation in experimental loadings with sideslip at high lift coefficients from the theoretically predicted trends are not known. However, at high angles of attack such as those under consideration, because of the transonic character of the flow (see reference 1), the Mach number of the flow over the upper surface of the airfoil is much higher than that of the free stream. It is possible, therefore, that the airfoil exhibits characteristics which the theory predicts for higher Mach numbers (bow wave attached), in this case a loss of positive dihedral effect. 1 (See reference 6.)

Mach lines swept behind the leading wing. - The variation of the experimental loading with angle of sideslip for both wings is presented in figure 5 for a Mach number of 1.70 and an angle of attack of 2.50. It is evident from a study of these data and similar data at 8.60 angle of attack (see fig. 6) that, even though the theoretical flow velocity component perpendicular to the edge of the leading wing was supersonic, the resulting load distributions differed little from those in the lower speed range where the flow component perpendicular to the leading wing was subsonic. At these higher supersonic speeds (M = 1.70) the value of tan e/tan \u03c4 for the leading wing panel was greater than 1.0 but less than the value for which the shock wave became attached to the sharp leading edge. Therefore, for both airfoils, transonic flow characteristics were evident in the form of a detached bow wave ahead of the swept leading edge. Since in the region between the detached bow wave and the leading edge the flow components perpendicular to the leading wing were subsonic, the flow around the leading edge of the airfoil was similar to the flow experienced when the Mach lines were swept ahead of the leading edge. It is not surprising, therefore, that the airfoils did not exhibit characteristics with regard to dihedral effect which are in agreement with the theory which assumes the bow wave to be

<sup>&</sup>lt;sup>1</sup>Verification of this hypothesis can be obtained by a separate examination of the pressure data on the upper and lower surfaces of the airfoil (tables II—VII). These data show that the distribution of loading on the upper surface of the wing is such as to result in the loss in positive dihedral effect.

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attached, but rather, because of these transonic effects, showed characteristics similar to those noted in the lower speed range (Mach lines swept ahead of the leading edge), that is, a positive dihedral effect at 2.5° angle of attack which was reduced to almost zero dihedral effect at 8.6° angle of attack.

### Comparison of Experimental and Theoretical Loading

A comparison of the theoretical load distribution with the experimental values at 9° of sideslip is presented in figure 7 for both the round-nose and the sharp-nose airfoil at a Mach number of 1.20. At the low angles of attack the data for the sharp-nose airfoil are in somewhat better agreement with the theory than the data for the round-nose airfoil. For the high angles of attack, the experimental loading for both airfoils revealed certain pressure discontinuities over the airfoil surface (as indicated on the trailing panel at about 70-percent semispan and on the leading panel at about 60-percent semispan), resulting in a reduction in the loading, which were quite similar to those noted in the data for triangular wings at zero sideslip (reference 1). The discontinuities over the triangular wing at zero sideslip have been shown to be associated with certain transonic effects (formation of shock waves) similar to those experienced over two-dimensional airfoils at transonic speeds (reference 3). It was concluded that the pressure discontinuities noted in the data of the present investigation also denoted the existence of shock waves on the airfoil surface and that the shock patterns in transverse planes resemble closely the patterns existing on triangular wings at zero sideslip (reference 1).

The theoretical and experimental load distributions at  $9^{\circ}$  of side—slip are compared for both airfoils in figure 8 for a Mach number of 1.70. Since  $\tan \varepsilon/\tan \mu$  of the leading wing is greater than unity, it is assumed in the theory that the bow wave is attached to the wing leading edge with the result that the lifting pressures are constant between the Mach line and the leading edge. In the actual case, how—ever, since the bow wave was detached, interaction between the upper and lower surface occurred, resulting in a pressure peak near the leading edge. Over the remaining portion of the span the experimental loading for both airfoils was in fairly good agreement with the theoretical.

#### CONCLUDING REMARKS

The present investigation of the characteristics of the pressure distribution over two triangular wings in sideslip for a Mach number

range of 1.20 to 1.70 was made to study the transonic effects on the load distribution over triangular wings in sideslip and to provide data for a comparison of the experimental and theoretical load distributions.

In the lower speed range (Mach lines swept ahead of the leading edge) the results of the investigation indicated at the high angles of attack a significant departure of the variation of the experimental loading with sideslip from that indicated by the linearized theory. The positive dihedral effect expected from theoretical considerations was in evidence at the low angles of attack but was not realized at the high angles of attack. The airfoils exhibited a variation of dihedral effect with lift coefficient similar to the variation with Mach number predicted by theory, that is, a loss of dihedral effect with increasing Mach number.

In the higher speed range where the Mach line was swept behind the edge of the leading wing panel, because of a detached bow wave ahead of the leading edge and resulting subsonic velocities at the leading edge, the load distributions exhibited characteristics similar to those experienced in the lower speed range wherein the Mach lines were swept ahead of the leading edge.

Ames Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Moffett Field, Calif.

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TABLE I.- AIRFOIL ORDINATES

# [Stations and Ordinates Given in Percent of Airfuil Chord]

Upper surface Lower surface  Station Ordinate Station Ordinate  0 0 0 0 0 1.25 .95 1.2595 2.50 1.31 2.50 -1.31 5.0 1.78 5.0 -1.78 7.5 2.10 7.5 -2.10 10 2.34 10 -2.34		Ordinate  0 .92 1.67 2.25 2.67		Ordinate  092 -1.67
0 0 0 0 0 0 1.2595 2.50 1.31 2.50 -1.31 5.0 1.78 5.0 -1.78 7.5 2.10 7.5 -2.10 10 2.34 10 -2.34	0 5 10 15 20	0 .92 1.67 2.25	0 5 10	092
1.25     .95     1.25    95       2.50     1.31     2.50     -1.31       5.0     1.78     5.0     -1.78       7.5     2.10     7.5     -2.10       10     2.34     10     -2.34	5 10 15 20	.92 1.67 2.25	5 10	92
15     2.67     15     -2.67       20     2.87     20     -2.87       25     2.97     25     -2.97       30     3.00     30     -3.00       40     2.90     40     -2.90       50     2.65     50     -2.65       60     2.28     60     -2.28       70     1.83     70     -1.83       80     1.31     80     -1.31       90     .72     90    72       95     .40     95    40       100     (.06)     100     (06)       100     0     0	30 40 50 60 70 80 85 90 95 100	2.92 3.00 2.94 2.75 2.45 2.02 1.47 1.15 .79 .40	20 25 30 40 50 60 70 80 85 90 95 100	-2.25 -2.67 -2.92 -3.00 -2.94 -2.75 -2.45 -1.47 -1.15 79 40



TABLE II.— EXPERIMENTAL PRESSURE COEFFICIENTS
[ M = 1.20]

										β =	00										
			0	7		e airfoi											ose air	foil			
Sta	ation	α	=-0.1°	α	=2.5°	a	=5.2°	α	=8.6°	α	=10.10	α	=0.0°	α	=2.7°	α	=5.2°	α	=8.8°	a	=10.1
x/cr	<u>w/2</u>	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	PZ	Pu	Pl	Pu	Pı
0.25	.933 .867 .800 .733 .667 .533 .400 .267	0.099 .033 .027 .029  .011 .012 .003 .002	0.544 .255 .085 .059 .043 .039 .034 .023 .015	 -0.080 139 114 096  079 067 064	0.654 .399 .228 .189 .158 .138 .121 .097 .088 .083	 -0.239. 302 269 230  154 143 124 115	0.717 .520 .356 .308 .267 .236 .206 .180 .167	 -0.454 511 461 433  269 235 209 191	0.747 .652 .512 .459 .412 .377 .336 .302 .286 .272	 -0.555 548 516 492  309 254 230 210	0.770 .710 .578 .524 .475 .437 .395 .360 .341 .326	 0.330 .251 .195 .157 .134 .099 .067 .052	0.373 .309 .255 .208 .178 .134 .094 .072	0.184 .141 .101 .070 .052 .028 0	 0.498 .423 .359 .308 .272 .217 .173 .148	 -0.221 006 011 026 038 047 064 064	 0.589 .507 .443 .391 .352 .294 .246 .219	 -0.665 436 311 256 176 140 151 149 148	0.686 .622 .560 .509 .471 .410 .360 .329	-0.382 -553 -417 -379 -324 -165 -174 -177	 0.73 .66 .61 .55 .53 .46 .41 .31
0.50	1 .967 .933 .900 .867 .800 .733 .667 .600 .533 .467 .400 .267 .133		.555 .069 026 058 058 076 096 106 101 093 087 073 074		.624 .257 .140 .091 .075 .033 -013 018 021 017 015 004 005		.634 .404 .285 .227 .199 .147 .112 .090 .080 .074 .073 .069	658 - 659 642 - 597 - 577 - 577 - 343 - 335 - 335 - 325 - 261	.581 .558 .443 .380 .347 .286 .245 .220 .209 .193 .193 .191 .236 .252		.588 .647 .545 .481 .448 .391 .327 .314 .310 .309 .339	.167 .094 025 047 091 111 129 131 123 111 090 079 075	.263 .184 .113 .066 .001 055 093 116 120 106 093 074 068		402 .307 .232 .180 .104 .0 0 026 032 023 014 .002 .005	565 - 359 - 185 - 202 - 254 - 280 - 285 - 280 - 268 - 247 - 215 - 194 - 187	.498 .397 .320 .265 .189 .130 .088 .065 .058 .062 .068 .085 .080	- 760 - 656 - 621 - 593 - 567 - 384 - 366 - 373 - 371 - 362 - 341 - 301 - 276 - 262	- 604 .499 .457 .402 .325 .266 .225 .201 .193 .199 .199 .215 .233 .246		
0.75	978 .978 .955 .933 .912 .889 .867 .845 .800 .756 .711 .667 .622 .578 .533 .446						 .215 .160 .114 .089 .067 .057 .043 .045  .077 .095 .113	717 - 721 - 708 - 700 - 691 - 679 - 637 - 635 - 621 - 561 - 402 - 386	 .438 .374 .341 .319 .301 .293 .285 .287  .299 .311 .316 .327	747 - 745 - 733 - 726 - 720 - 713 - 682 - 667 - 649 - 531 - 467	.561 .512 .453 .390 .369 .357 .343 .343 .355 .355 .359		035 023 080 127 161 182 199 204 180 156 137 129	156 - 135 - 206 - 242 - 271 309 - 309 - 310 - 290 - 272 - 272 - 218	 .172 .104 .043 014 046 067 076 076 073 043 041	- 566 - 527 - 515 - 494 - 476 379 - 379 - 359 - 337 - 384		- 744 - 695 - 679 - 662 - 659 - 650 573 - 465 - 442 - 428 - 407	 .476 .412 .377 .330 .309 .296 .292 .278 .281 .287  .288 .294	864 840 807 774 758 722 686 594 499 456	 .52 .47 .42 .39 .37 .36 .35 .33 .33 .33

# TABLE II. - CONTINUED

									R	ight win	g panel										
						nose airi			0				0 1			narp-nos			0		0
Stat		α=	-0.1°	α=	2.5°		5.2°	α=	8.6°		10.10		0.00		2.70		5.2°		8.8°		10.10
x/cr	<u>y</u>	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl
0.25	.933 .867 .800 .733 .667 .533 .400 .267	0.126 .039 .027 .028  .003 009 006	0.698 .316 .118 .078 .053 .044 .039 .035 .016	 -0.042 132 126 110  087 087 074 066	0.767 .463 .265 .216 .179 .155 .131 .106 .093 .088	 -0.185 275 286 256  163 149 128 110	0.842 .603 .404 .347 .301 .267 .232 .197 .180	 -0.395 467 460 437  322 248 216 174	0.870 .720 .546 .484 .433 .391 .344 .302 .284		0.941 .825 .655 .588 .532 .489 .439 .393 .371	0.385 .300 .231 .184 .154 .113 .073 .054	 0.446 .375 .310 .250 .212 .157 .108 .078	0.243 .179 .123 .089 .066 .038 .004 009	0.579 .485 .411 .348 .305 .239 .184 .153	 -0.226 .047 .015 019 032 042 067 072 069	0.685 .585 .507 .441 .398 .325 .267 .231	 -0.643 474 296 131 129 144 154 151 143	0.808 .701 .622 .560 .514 .439 .378 .339	-0.715 587 499 236 168 174 180 172 162	 0.84 .74 .67 .61 .57 .50 .43 .39
0.50	1 .967 .933 .900 .867 .800 .733 .667 .600 .533 .467 .400 .267 .133	 010 071  088 114 123 139  129 123 122 105 089	.678 .136 .014 024 059 059 099 099 099 093 088 088	 161 267  272 246 240 249 201 201 177 158	.761 .325 .185 .128 .108 .059 .021 001 008 014 010 010		.784 .485 .345 .268 .237 .177 .134 .111 .100 .089 .090 .090	585 - 611 601 - 582 - 563 - 529 - 462 - 330 - 341 - 309 - 272	.739 .623 .486 .417 .378 .309 .229 .218 .303 .199 .197 .241 .262	702 - 702 - 682 - 669 - 654 - 644 - 607 565 - 375 - 373 - 327 - 287	. 784 . 756 . 624 . 545 . 506 . 439 . 367 . 339 . 343 . 340  . 359 . 378	 .228 .139 015 080 107 131 140 135 120 095 079	349 .257 .176 .118 .040028076109122114100070059053		.496 .386 .293 .229 .141 .070 .019 017 022 012 .010 .017		.545 .482 .391 .323 .158 .108 .078 .078 .074 .081 .092 .095 .103	695 - 584 - 540 - 516 - 506 - 351 - 374 - 377 - 373 - 366 - 271 - 253	. 708 . 604 . 527 . 463 . 371 . 300 . 251 . 223 . 214 . 214 . 238 . 253 . 271	803 - 677 - 643 - 621 - 593 - 194 - 393 - 395 - 316 - 282 - 265	.777. .684 .660 .54 .46 .399 .35 .33 .32 .333 .344 .344
0.75	1 .978 .955 .933 .912 .889 .867 .845 .800 .756 .711 .667 .622 .578	125154179196222234224209196179169			118 .054 .007023047057066063056037018	478 - 476 - 482 - 485 498 - 505 - 490 - 454 - 433 - 403 - 366 - 334	.266 .182 .149 .120 .094 .083 .067 .064  .083 .105 .124	697 - 680 - 675 - 667 - 658 - 648 - 636 - 617 - 595 - 571 - 527 - 507	 .474 .418 .363 .337 .315 .306 .295 .298  .305 .317 .335 .353		.583 .495 .466 .439 .416 .403 .388 .390 .388 .391 .404		124 .044023075118145172191184167143130107		.254 .162 .094 .032 011 039 065 075 059 042 036	490 - 449 - 448 - 446 - 448 402 - 384 - 394 - 385 - 368 - 341 - 299	 ·353 ·264 ·196 ·139 ·105 ·082 ·073 ·063 ·076  ·078 ·104 ·214			- 783 - 733 - 723 - 704 - 698 - 691 - 667 - 645 - 609 - 475 - 407	.61 .53 .47 .43 .41 .39 .38 .36 .37 



TABLE II. - CONTINUED

										β = 5 <sup>c</sup>											
									I	eft wing	panel	1									
			-0.1°			-nose ai			0			-	0			arp-nose					
Sta	ition	α=	-0.1	0	=2.5°	α	=5.2°	0	=8.6°	0	=10.10	0	=0.0°	(	1=2.7°	α=	=5.2°	α=	-8.8°	α	10.10
x/cr	<u>w/2</u>	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl
0.25	933 .867 .800 .733 .667 .533 .400 .267	0.097 .038 .037 .036  .016 .006 .011	0.427 .194 .060 .044 .031 .028 .025 .014 .011	 -0.080 120 093 077  064 065 055 063	0.545 .377 .195 .166 .139 .123 .108 .091 .083	 -0.280 306 243 208  147 138 123 126	0.592 .447 .309 .270 .234 .210 .188 .166 .156	 -0.510 501 454 442  227 223 195 194	0.616 .579 .463 .420 .380 .350 .317 .290 .277 .264	 -0.572 560 529 512  267 248 225	0.622 .622 .578 .475 .434 .402 .367 .339 .325	0.286 .224 .178 .146 .125 .096 .069 .062	0.286 .249 .209 .169 .149 .113 .080 .063	0.151 .112 .077 .057 .043 .023 0	0.417 .345 .297 .253 .229 .188 .152 .131	 -0.254 049 021 030 038 048 062 062	0.525 .447 .402 .346 .318 .271 .231 .209	 -0.527 491 419 335 213 117 144 143	0.594 .551 .503 .460 .432 .385 .344 .320	 -0.571 573 511 452 366 129 171 172 186	 0.644 .589 .544 .503 .476 .428 .389 .365 .337
0.50	.967 .933 .900 .867 .800 .733 .667 .600 .533 .467 .400 .267 .133		.436 .006 057 078 075 108 104 101 094 089	265 - 298 - 260 - 238 - 220 - 208 - 202 185 - 176 - 176 - 145	.500 .200 .106 .066 .050 .018 020 021 023 020 019 009	- 476 - 508 - 477 - 451 - 400 - 304 - 294 255 - 234 - 241 - 222 - 210	.495 .339 .238 .185 .161 .118 .088 .070 .062 .057 .057 .055 -065	- 720 - 699 - 682 - 667 - 590 - 392 - 338 - 320 - 319 - 265	.433 .493 .403 .347 .317 .266 .232 .211 .203 .194 .200 .193 .238 .241	744 - 719 - 716 - 703 - 685 - 664 - 592 - 370 - 354 - 349 - 317 - 289	.427 .565 .485 .432 .403 .355 .325 .311 .304 .293 .293 .288		170 .109 .049 .014036080108121117105095078072		-312 .228 .160 .110 .061 .011 020 036 037 028 021 001		. 430 . 344 . 268 . 223 . 160 . 109 . 074 . 056 . 050 . 055 . 060 . 069	- 779 - 761 - 609 - 516 - 536 - 494 - 386 - 345 - 330 - 314 - 290 - 271	-525 -457 -405 -362 -299 -251 -219 -201 -199 -197 -203 -215 -232 -222		-357 -574 -517 .469 .432 .378 .335 .307 .294 .287 .291 .291 .306
0.75	.867 .845 .800				 .028 .016 047 062 076 078 082 074 		.165 .121 .075 .057 .039 .032 .025 .033 .076 .104 .152 .198	764 - 761 - 7752 732 713 - 696 - 678 - 644 - 425 - 414 - 374 - 367	.401 .356 .328 .301 .285 .279 .272 .274  .283 .291 .295 .308	773 - 776 - 761 - 753 742 - 732 - 732 - 698 - 608 - 508 - 450 - 419	.446 .411 .365 .347 .329 .321 .309 .314 	007 -030 -112 -146 -169199 -201 -172 -158 -141 -126 -102	 039 092 139 192 200 211 206 194 166 149 137 132 108	202 - 185 - 245 - 271 - 315 - 315 - 315 - 275 - 249 - 233 - 224 - 213 - 202			.239 .166 .116 .078 .056 .048 .037 .052 .071		.421 .371 .335 .305 .287 .279 .279 .264 .272 .272 .277 .286	- 856 - 841 - 828 - 775 - 776 - 646 - 568 - 538 - 538 - 361	



TABLE II.- CONTINUED

									Ri	ght win	g panel			100							
					Round-n	ose air	foil									harp-no	se airf	oil			
Sta	tion	α=	-0.1°	α=	2.5°	α=	5.2°	α=	8.6°	α=	10.10	α=	0.0	α=	2.7°	α=	5.2°	α=	8.8°	α=	10.10
x/cr	¥/2	Pu	P <sub>7</sub>	Pu	P <sub>7</sub>	Pu	Pz	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl
0.25	1 •933 •867 •800 •733 •667 •533 •400 •267	0.163 .049 .028 .027  0 015 011	0.736 .460 .146 .094 .062 .048 .041 .027 .016	 -0.007 112 129 121  097 080 046	0.847 .508 .293 .237 .194 .166 .138 .108 .093				0.976 .774 .582 .511 .453 .407 .355 .309 .285			0.425 .337 .258 .204 .167 .118 .074 .053 .034	0.506 .424 .350 .282 .235 .169 .112 .079	0.296 .219 .151 .110 .081 .051 .012 004 012	0.661 .550 .467 .393 .346 .270 .203 .165				0.863 .761 .672 .602 .550 .466 .394 .348		
0.50	1 .967 .933 .900 .867 .800 .733 .667 .600 .533 .467 .407 .207	075056075107125149132132131094	.778 .188 .051 0 009 072 096 099 101 095 091 063 044	104 226 255 260 242 254 231 215 208 160	.862 .370 .217 .156 .133 .077 .033 .006 002 008 007 006			- 494 - 525 540 - 539 - 534 - 531 - 469 - 435 - 354 - 299 - 257	.861 .676 .533 .451 .410 .335 .281 .249 .223 .223 .224  .266 .283			- 274 .176 - 085 .011 - 061 - 102 - 131 - 146 - 146 - 143 - 102 - 082 - 072	414 .314 .224 .159 .071005059120117105072055048	 .105 .047 039 099 158 175 209 214 213 195 160 134 119	 .579 .458 .350 .279 .180 .098 .042 002 021 017 007 .016 .027 .031				.787 .675 .579 .507 .401 .327 .243 .233 .233 .237 .243 .270 .293 .315		
0.75	1 .978 .955 .933 .912 .889 .867 .845 .800 .756 .711 .667 .622 .578	 096 133 160 180  211 233 242 238 223 223 2193 193 193 182	017 049 083 113 142 157 169 169 155 133		163 .093 .042 .007025038051051049035			634 - 611 - 614 - 607 604 - 603 - 600 - 593 - 579 - 516 - 516				140 .079011066106168198218216214205186	.183 .095 .022 035 083 116 148 170 173 164	057064130179215268292306297296286	.316 .219 .143 .074 .027 006 040 054 041 025				.584 .495 .433 .382 .352 .333 .324 .316 .315 .317		



TABLE II .- CONCLUDED

										β =	00										
									Le	ft wing											
				Ro	ound-nos	e airfo	il								Sh	arp-nos	e airfo	il			
Stat	tion	o	u=-0.1°	α=	=2.5°	α=	5.2°	α=	8.6°	α=	10.1°	α=	0.0	α=	2.70	α=	5.20	α=	8.8°	α=	10.1
x/cr	<u>m/5</u>	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu.	Pl	Pu	Pl	Pu	Pl	Pu	Pl
0.25	1 .933 .867 .800 .733 .667 .533 .400 .267	0.081 .031 .031 .031 .012 .002 .007 010	0.343 .153 .037 .028 .018 .015 .010 .005 004 018	 -0.090 112 084 072 065 067 059 076	0.447 .277 .161 .138 .114 .102 .088 .073 .066				0.492 .505 .416 .383 .347 .323 .294 .272 .264			0.240 195 154 128 112 .085 .063 .050	0.255 .194 .171 .138 .124 .093 .065 .058	 0.104 .084 .058 .042 .030 .011 009 011	0.333 .283 .244 .211 .191 .154 .124 .109				 0.525 .484 .446 .413 .393 .354 .321 .303 .273		
0.50	1 .967 .933 .900 .8667 .800 .733 .667 .600 .533 .467 .400 .267 .133 0		.322 040 086 098 095 102 111 115 109 106 100 097 		.392 .144 .067 .036 .003 003 031 032 033 030 029 021 028			- 755 - 728 - 723 - 710 - 677 - 456 - 364 - 332 - 313 - 289 - 268	.300 .424 .354 .305 .279 .237 .208 .191 .194 .176 .180 .184 .205						.229 .160 .109 .077 .028 .011 .035 .045 .045 .034 .019 .007				 .450 .392 .347 .313 .264 .225 .200 .193 .185 .187 .203 .208 .191		
0.75	1 .978 .955 .933 .912 .889 .867 .845 .800 .756 .711 .667 .578 .533 .446	150166180186186177163149136120110						798 - 801 - 782 - 771 - 773 - 753 - 283 - 462 - 434 - 360 - 356	 .351 .316 .280 .267 .254 .248 .242 .246  .254 .263 .267 .278						.046 003 042 080 080 095 092 067 066 067 066 047			- 743 - 756 - 769 - 718 - 672 - 629 - 513 - 475 - 372 - 327 - 326			



# TABLE III.— EXPERIMENTAL PRESSURE COEFFICIENTS [ M = 1.30 ]

				Pour	d-nose a	irfoil				β = (	00				- 8	harro-nos	e airfo	11			
Stat	ton	α=-0	0.10		2.50		5.20	α=6	.60	α=10	0.10	α=0	0.00	α=2.		α=5		α=8.	.80	α=10	10
x/cr	<u>A</u> /5	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pı
0.25	1 •933 •867 •800 •733 •667 •533 •400 •267	0.113 .036 .013 .022  .007 .003 .004 .013	0.551 .216 .088 .055 .026 .032 .028 .024 .020	 -0.022 111 115 093  071 067 059 044	0.654 .353 .205 .170 .117 .115 .107 .094 .086	-0.164 250 252 221 146 129 115 095	0.721 .474 .337 .292 .237 .228 .201 .181 .167	-0.358 412 420 387 274 216 193 164	0.749 .608 .484 .437 .376 .355 .317 .292 .277 .268	-0.421 481 472 440 330 229 211	0.747 .641 .529 .480 .426 .398 .355 .328 .312 .303	0.352 .261 .201 .162 .136 .101 .072 .064	 0.374 .315 .261 .213 .181 .137 .098 .077 .076	0.193 .153 .105 .073 .052 .030 .009 .005	0.488 .411 .351 .301 .268 .218 .175 .153	-0.161 .041 .003 019 031 039 054 044 049	0.583 .495 .431 .380 .347 .293 .248 .222	-0.572 405 266 114 121 118 127 114 105	0.684 .618 .555 .503 .469 .413 .365 .338 .316	-0.638 -513 -426 -223 -163 -158 -164 -151 -143	
0.50	1 .967 .933 .900 .867 .800 .733 .667 .600 .533 .467 .400 .267	059031055077083099091093089088077067	.549 .117 016 024 040 065 078 079 077 073 070 062 053 054	 116 203  202 183 177 183 173 162 153 148 133 118	.614 .284 .176 .120 .100 .063 .029 .010 .006 .004 .012 .002 .007	 274 346  356 343 307 275 252 234 218 209 188 173 168	.624 .422 .309 .243 .219 .168 .126 .105 .094 .089 .081 .078 .082		.608 .566 .454 .393 .357 .298 .253 .229 .213 .209 .196 .197 .183 .187	 -586 -576  -573 -5497 -497 -494 -293 -288 -273 -245 -250	.588 .604 .503 .437 .401 .343 .297 .273 .258 .249 .237 .234 .222 .231 .240	235 .162 .104 .076 .002 - 041 060 079 088 087 078 059 048	306 - 233 - 160 - 115 - 049 - 044 - 071 - 082 - 076 - 065 - 044 - 034 - 027		436 .343 .267 .216 .145 .038 .007003 .003 .009 .022 .029		 .526 .427 .351 .299 .223 .159 .112 .081 .070 .074 .082 .093 .087	 606 504 464 447 436 363 279 301 298 286 267 233 211 199	 .627 .537 .472 .419 .344 .282 .238 .211 .199 .199 .200 .200	661 - 574 - 540 - 522 - 510 - 501 - 410 - 325 - 326 - 320 - 268 - 244 - 232	.657 .570 .506 .453 .377 .274 .248 .234 .233 .240 .230
0.75	1 •978 •955 •933 •912 •889 •867 •845 •800 •756 •711 •667 •662 •578 •533 •446		043074128149151130124115		.097 .046 .014 .035 .040 .044 .043 .054 .054 .046	428 - 421 - 417 - 416 427 - 423 - 400 - 372 - 351 - 3287 - 272	225 .170 - 106 .080 .071 .054 .055 040 .042 .049		.370 .314 243 .220 .207 .187 .171 .171 .168 .179 .213 .304		.417 .357 .293 .272 .261 .247 .255 .315 .314 .363 .374	-112 .062 -023 063 063 07 141 159 164 155 147 136 127	 .098 .031 023 068 101 124 144 159 147 136  115 107		 .220 .142 .086 .031 002 027 048 061 041 029 		 .305 .229 .171 .119 .087 .068 .058 .044 .057 .071 		 .435 .367 .315 .270 .243 .225 .213 .193 .200 .204  .184 .193 .273	639 - 610 - 587 578 581 - 578 560 - 550 - 438 - 374 - 374	.47 .40 .35 .31 .29 .27 .26 .25 .26 .30

TABLE III .- CONTINUED

-										β =	_										
-				Por	nd_noge	airfoil				Right win	ng panel	T -				~1·					
Sta	tion	α=-	0.10		2.50		.20	α=8.	60	α=10	0.10	α=0	.00	α=	2.70	Snarp- α=5	nose ai	rroil α=8	.8°	α=10	.10
x/c <sub>r</sub>	y w/2	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pı	Pu	Pz	Pu	P
0.25	.933 .867 .800 .733 .667 .533 .400 .267	0.167 .058 .017 .025  .008 .001 .001	.128 .080 .035 .037 .033 .028	102  077 073	0.769 .418 .256 .206 .140 .138 .126 .109 .097		0.830 .533 .380 .277 .261 .247 .213 .189 .172	 -0.275 350 378 372  293 240 202 167	0.879 .680 .528 .470 .397 .374 .327 .294 .272 .260	 -0.347 427 444 426  339 271 219 184	0.870 .708 .572 .513 .449 .418 .368 .334 .312 .298	 0.387 .305 .234 .189 .157 .115 .076 .057	 0.450 .374 .311 .250 .215 .161 .109 .078	0 . 246 . 187 . 129 . 092 . 065 . 007 003 009	 0.572 .469 .398 .336 .297 .237 .184 .151	 -0.148 .115 .036 .001 018 034 054 059 060	0.683 .573 .496 .432 .387 .319 .261 .227	 -0.500 363 275 098 107 126 135 131 120	0.768 .619 .601 .539 .495 .423 .363 .325 .295	428 372 300 153 158 166 161	0 · 306 · 719 · 646 · 586 · 404 · 365 · 333
0.50	1 .967 .933 .900 .867 .800 .733 .667 .600 .533 .467 .400 .267	 .114 001 	.071 .016 .001 019 050	190 200 190 199 192 182 171 164 144	.749 349 .213 .163 .138 .092 .047 .002 .006 .007 .007 .016 .016		. 765 . 484 . 346 . 286 . 253 . 193 . 116 . 102 . 097 . 086 . 082 . 087 . 088		.772 .639 .508 .434 .393 .268 .238 .219 .206 .201 .189 .188		.743 .672 .548 .477 .437 .365 .211 .221 .226 .236		396 .307 .217 .161 .085 .014035070088087079075042030	112 .062 .005 014 082 119 137 162 168 159 135 115 104	526 .417 .315 .255 .173 .099 .048 .010 011 001 004 .020		.627 .517 .423 .357 .267 .192 .137 .095 .074 .079 .088 .092 .098		 .714 .613 .525 .459 .368 .273 .238 .201 .184 .185 .187 .186	471 454 455 453 448 335 337 338 322 280 242	761 . 659 . 573 . 510 . 416 . 340 . 287 . 253 . 236 . 235 . 238 . 238 . 238
0.75	1 .978 .955 .933 .912 .889 .867 .860 .756 .711 .667 .622 .578 .533 .446	 052 083 108 125  173 182 178 165 161 146	 .020 049  090 113 124 135 145  127 125 115 115	233 255 268  286 294 294	 .158 .090 		 -276 .218  .139 .107 .095 .074 .071  .049 .050 .054		 .425 .359 273 .246 .231 .207 .196 .180 165 .163 .173 .260		.457 .394 .293 .279 .256 .246 .237 .246 .278 .308		170 .092 .030022063092119138136129121		030 052 044 039  035 031	330 - 298 - 311 - 312 - 330 298 - 300 329 - 315 - 310 - 298 - 315 - 326			 .489 .407 .337 .290 .255 .233 .217 .196 .198  .185 .182		.55% .46% .40% .318 .29% .28% .25% .25% .25% .25% .25% .33%

# TABLE III .- CONTINUED

										β =										,	
				D		airfoi	7		I	eft win	g panel					The same of		0-43			
Sta	ation	α=-0	.10	α=2		α=5		α=8	.6°	α=1	0.10	α=0	.00	α=2		harp-no α=5	.2°		3.8°	α=1	.0.1°
x/c <sub>r</sub>	<u>y</u>	Pu	Pı	Pu	Pı	Pu	P <sub>1</sub>	Pu	Pı	Pu	Pı	Pu	Pı	Pu	Pl	Pu	Pı	Pu	Pl	Pu	Pı
0.25	1 .933 .867 .800 .733 .667 .533 .400 .267	0.107 .036 .021 .031  .018 .014 .015 .018	0.450 .157 .052 .036 .030 .019 .023 .019 .017	 -0.050 108 097 075  058 054 048	0.599 .288 .173 .150 .108 .100 .090 .083 .078	 -0.219 268 230 191  129 116 105 097	0.589 .400 .293 .257 .213 .206 .185 .169 .160	 -0.422 442 396 372  224 199 180 164	0.609 .530 .429 .392 .343 .322 .297 .276 .266	 -0.486 521 481 454  258 219 207 192	0.616 .581 .484 .444 .397 .375 .344 .323 .311 .304	0.279 .224 .174 .142 .122 .092 .071 .067	 0.279 .248 .205 .168 .145 .108 .079 .063	0.151 .111 .076 .053 .039 .019 .004 .002	 0.402 .335 .286 .245 .222 .180 .145 .127	 -0.123 005 017 029 039 047 058 053 060	 0.486 .425 .372 .330 .301 .256 .221 .199 .181	 -0.566 408 277 231 153 126 133 124 127	 0.581 .530 .482 .443 .417 .372 .337 .315 .295	 -0.610 506 385 354 273 150 160 153 159	0.616 .567 .521 .482 .456 .415 .371 .355
0.50	1 .967 .933 .900 .867 .800 .733 .667 .600 .533 .467 .400 .267 .133		.394 .041 028 047 049 058 075 082 079 076 065 062 056	175 - 228 185 - 174 - 163 - 155 - 146 - 138 - 138 - 123 - 123 - 112	.464 .219 .128 .088 .071 .041 .002 001 .004 0 .007 .001 .006 001	 -358 -408 	.465 .346 .248 .203 .179 .138 .104 .088 .081 .071 .069 .073		.544 .490 .396 .314 .267 .231 .212 .205 .190 .190 .174 .180	657 - 644 599 - 568 - 553 - 525 - 419 - 308 - 292 - 289 - 264 - 244 - 254	.416 .528 .443 .395 .365 .278 .258 .246 .234 .238 .225 .219 .227	 .173 .111 .073 .049 -015 045 077 077 071 062 048 040		014026054064114131148148148149122105096095	 .340 .258 .189 .150 .089 .008 -011 014 007 001 .016 .014		 .432 .354 .287 .245 .180 .126 .090 .067 .062 .068 .070 .078 .079		 .521 .451 .397 .358 .297 .247 .212 .193 .186 .188 .185 .181 .182	261 242	
0.75	1 .978 .955 .933 .912 .889 .867 .845 .800 .756 .711 .667 .622 .578 .533				 .128 .007041055058054066052041022	459 - 456 - 450 - 446 439 - 409 - 386 - 340 - 264 - 242 - 233			 .318 .275  .212 .194 .183 .172 .173 .171  .195 .236 .270	684 - 680 - 673 - 662 - 655 - 639 - 6613 - 577 - 452 351	389 .326 291 .286 .281 .284 .288 .295 332 .335 .344 .352	064 .021052087114151159143132122115107					251 .173 .125 .082 .059 .048 .043 .046 .055 044 .045		368 309 266 230 209 197 190 167 187 186 	642 625 624 617 563 447 401 375	



# TABLE III. - CONTINUED

											= 90			dia i							
			-	D	ound-nos	10 01 100				Right W	ing panel	1									
Sta	tion	α=-0	.10	α=2		α=5.		α=8.	6°	α=10	).1°	α=0.	.0°	α=2.		arp—nos	e airfo:	il α=8.	80	α=10	0.10
x/c <sub>r</sub>	<u>y</u>	Pu	Pl	Pu	Pl	Pu	Pl	Pu	PZ	Pu	Pl	Pu	Pl	Pu	Pl	Pu	PZ	Pu	Pl	Pu	PZ
0.25	1 •933 .867 .800 •733 •667 •533 •400 •267	 0.203 .085 .029 .031  .009 002 003	.163 .105 .049 .041 .033 .029	0.088 051 104 099  086 082	0.871 .486 .290 .230 .154 .149 .134 .114 .098			-0.211 297 338 339 312 257 210 166	0.980 •737 •562 •495 •415 •391 •336 •296 •269 •252			 0.436 .344 .263 .209 .170 .121 .076 .049	 0.505 .425 .352 .284 .239 .174 .116 .077	 0.288 .228 .160 .113 .077 .036 .002 016 021	0.630 .529 .446 .372 .325 .251 .188 .148			 -0.407 301 228 112 092 132 141 143 127	0.830 .732 .641 .571 .517 .434 .363 .319 .285		
0.50	.467 .400 .267		.053 .026 002 036 060 070 075 074 070	.004 116  157 180 190 205 200 193 183 174 152 129	.859 .379 .231 .198 .170 .119 .067 .035 .019 .009 .007 .007				.890 .692 .542 .465 .420 .343 .280 .246 .224 .207 .198 .192 .188 .180			339 .244 .179 .143 .059014050089106112090063045	468 .371 .273 .209 .124 .043 014 059 086 091 087 060 038 026	.161 .104 .052 .019 055 106 129 163 173 182 176 151 116	.591 .474 .374 .306 .209 .126 .068 .019 008 015 014 .011			456 - 370 - 355 - 345 - 354 - 366 - 333 - 298 - 315 - 318 - 307 - 266 - 218 - 196	 .788 .672 .575 .502 .399 .313 .251 .206 .184 .181 .181 .190		
0.75	,933 ,912 ,889 ,867 ,845 ,800 ,756 ,711 ,667		007 054 082 096 113 117 132 	213 229  255 271	.209 .135 .058 .026 .010 006 017 039 037 032 006				.468 .393 .298 .268 .251 .226 .211 .192 .171 .167 .169 .208									455 - 421 - 430 445 - 458 - 463 - 458 - 450 - 433 - 371 - 367	 .547 .450 .379 .316 .270 .247 .206 .201 .201  .191 .186 .174		

# TABLE III. - CONCLUDED

										β =	90										
-	_			-				-		Left win	g panel		_								
						e airfoi											se airfo				
Stati	on	α=-	0.1°	α=2.	5°	α=5.	2°	α=8.	.6°	α=1	0.10	α=0	.0°	α=2	.7°	α=5	.2°	α=8.	8°	α=10	0.10
x/c <sub>r</sub>	y w/2	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	PZ
0.25	1 .933 .867 .800 .733 .667 .533 .400 .267	 0.088 .030 .018 .028  .014 .010 .011	0.359 .113 .028 .021 .001 .010 .012 .010 .007 001	 -0.061 101 087 067  055 054 049 053	0.454 .236 .146 .126 .090 .092 .086 .078 .072			-0.456 451 402 385 208 193 179	0.484 .457 .390 .360 .316 .300 .279 .264 .255 .246			 0.230 .187 .146 .120 .102 .075 .060 .059	0.214 .194 .160 .131 .113 .084 .060 .049	0.128 .091 .064 .046 .036 .018 .004 .005	0.342 .284 .244 .212 .193 .159 .131 .118			-0.488 452 367 289 171 120 133 126 146	0.502 .462 .426 .394 .376 .342 .313 .297 .275		
0.50	1 .967 .933 .900 .867 .800 .733 .667 .600 .533 .467 .400 .267 .133		.256 006 058 072 071 086 083 077 069 065 074	 205 220 181 156 156 149 192 133 131 120 110	.357 .163 .086 .058 .043 .018 001 010 008 006 009 002 007 004			604 - 585 - 559 - 559 - 559 - 294 - 279 - 261 - 257 - 238 - 225 - 236	.294 .407 .336 .305 .280 .239 .208 .191 .181 .177 .179 .177 .166 .171	· · · · · · · · · · · · · · · · · · ·		133 .074 .042 .023 -030 -0571 -078 -079 -075 -066 -052 -043	133 .084 .038 .011034066090100098087060048054		.271 .202 .139 .056 .017 006 020 019 013 007 .007				432 .378 .336 .308 .258 .219 .193 .180 .171 .175 .173 .172		
0.75	1 .978 .955 .933 .912 .889 .867 .845 .800 .756 .711 .667 .622 .578 .533								288 .255 201 .185 .180 .175 .182 .188 238 .251 .264			027008073100120149152131121114107097			088 .030011046066071045048038				.321 .273 .239 .212 .196 .189 .188 .171 .207 .228		



TABLE IV.— EXPERIMENTAL PRESSURE COEFFICIENTS
[M = 1.40]

										β	= 0										
				Rot	und-nose	airfoi.	1									Sharp-no	se airf	oil			
Sta	tion	α=-	0.10	α=2.5	50	α=5	.2°	α=8.	6°	α=10	.ı°	α=0	.0°	α=2	.7°	α=5.	20	α=8.8	80	α=1	.0.1°
x/cr	y w/2	Pu	Pl	Pu	Pı	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl
0.25	.933 .867 .800 .733 .667 .533 .400 .267	0.133 .055 .033 .031  .013 .004 .011	0.553 .271 .107 .065 .039 .029 .030 .025 .020	0.003 071 086 079  059 058 045 039	0.653 .388 .214 .169 .138 .121 .110 .092 .083 .084	 -0.124 187 198 185  129 116 096 088	0.698 .483 .318 .271 .236 .211 .187 .165 .151	 -0.259 304 317 321 247 203 168 149	0.763 .615 .472 .419 .372 .338 .302 .270 .255 .246	-0.320 -376 -383 -373 292 232 186 172	 .653 .517 .463 .416 .391 .343 .319 .295 .283	 0.340 .264 .205 .163 .126 .084 .054 .042	0.367 .326 .259 .197 .169 .118 .080 .055	 0.201 .134 .089 .053 .023 009 030 035 042	 0.480 .394 .322 .256 .224 .168 .125 .095	 -0.065 .079 .042 .016 016 043 056 056	0.578 .510 .436 .371 .339 .280 .230 .196	 -0.440 310 239 075 110 133 137 132 122	 0.685 .621 .546 .484 .455 .396 .344 .309	-0.507 -394 -325 249 135 155 158 151 143	0.707 .649 .578 .517 .490 .431 .380 .347
0.50	1 .967 .933 .900 .867 .800 .733 .667 .600 .533 .467 .400 .267 .133		.602 .139 .032 003 013 051 069 071 065 062 044 045		.668 .283 .172 .124 .108 .068 .034 .014 .007 .001 .004 .005		.678 .402 .284 .230 .207 .156 .117 .095 .084 .075 .072	-367 -388 -388 -390 -379 -343 -286 -261 -229 -203	.662 .555 .437 .381 .348 .288 .241 .211 .200 .182 .179 .173		.653 .593 .482 .423 .391 .329 .255 .255 .246 .227 .223 .217		317 .254 .185 .130 .058 011 047 089 083 077 056 043 051	 .077 .030 		268 -170 -087 -091 -169 -183 -190 -200 -211 -201 -193 -162 -136	 .519 .439 .372 .316 .234 .158 .117 .067 .082 .071 .079 .085 .083			5-3 458 448 432 430 420 381 296 310 301 290 254 225 222	 .647 .567 .519 .463 .382 .309 .261 .230 .221 .219 .228 .221 .219
0.75	1 .978 .955 .933 .912 .889 .867 .845 .800 .756 .711 .667 .622 .578 .533 .446		 .006 026 068 080 107 115 124 124 124 115 110 103 081		129 .077 .037 .009 014 025 035 035 035 037 040		 .232 .167 .135 .107 .083 .073 .056 .054  .040 .041		 -388 .337 .281 .253 .227 .213 .188 .180  .163 .168		434 .370 .325 .298 .271 .258 .233 .226205 .207 .206		131 .067 .022 029 065 015 137 131 129 120			302 277 280 294 283 250 273 285 281 278 271 250	 .340 .262 .207 .113 .079 .064 .041 .053 .075  .068	473 443 445 454 454 454 455 353 351 334 338	 .458 .381 .329 .273 .235 .212 .206 .184 .206 .215	541 505 515 497 497 495 481 465 439 367 342	 .489 .425 .377 .323 .287 .267 .262 .217 .260 .261



										β = Right wi		1				_					
				Ro	ound-nose	airfoil	L				-01					Sharp-no	se airf	oil	3.7		
St	ation	α=-	0.10	α=2.	.5°	α=5.2	20	α=8.	.6°	α=10	.1°	α=0	.0°	α=2	.7°	α=5	.2°	α=8.	8°	α=10	.10
r/c <sub>r</sub>	<u>प्र</u> ₩/2	Pu	Pı	Pu	Pl	Pu	Pl	Pu	PZ	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl
0.25	1 .933 .867 .800 .733 .667 .533 .400 .267	 0.176 .083 .044 .031  .005 009 .004	0.664 .344 .156 .099 .060 .035 .028 .023 .015	 0.062 030 069 079  074 072 058 047	0.764 .462 .262 .200 .159 .133 .155 .096 .082	 -0.056 143 177 178  149 133 109 096	0.818 .566 .359 .299 .255 .225 .196 .167 .150		0.891 .693 .512 .447 .394 .354 .312 .272 .252 .238	 -0.259 329 355 349  303 258 200 178	0.888 .725 .561 .497 .443 .399 .355 .313 .291	 0.415 .323 .253 .199 .153 .098 .060 .040	 0.453 .398 .320 .245 .209 .135 .096 .059	 0.282 .209 .151 .105 .066 .020 006 018 021	0.579 .496 .406 .329 .293 .223 .167 .125	 0.002 .124 .074 .039 002 045 064 068	 0.652 .573 .486 .411 .373 .302 .242 .198 .169	 -0.331 225 183 118 082 132 139 138 127	 0.783 .699 .609 .533 .496 .421 .357 .314 .276	 -0.417 302 257 226 170 153 164 159 146	0 .808 .724 .636 .562 .524 .451 .388 .346
0.50	1 .967 .933 .900 .867 .800 .733 .667 .600 .533 .467 .400 .267 .133	151 .028008040058	.735 .218 .095 .093 .026 006 033 054 062 069 067 065 045		.813 .354 .222 .167 .1146 .101 .059 .029 .014 .006 .004 .005 .013	-123 -225 -247 -250 -259 -264 -270 -232 -216 -163	.831 .474 .339 .273 .257 .184 .141 .110 .095 .081 .079 .076	- 271 - 371 - 337 - 357 - 370 - 345 - 325 - 325 - 325 - 239 - 203	.820 .625 .494 .421 .385 .315 .261 .226 .210 .188 .186 .173		.801 .663 .535 .469 .431 .359 .306 .272 .253 .233 .228 .216	315 .224 131 .034 009 041 072 090 097 095 057 050		-154 -102 -016 -059 -097 -121 -146 -158 -161 -159 -132 -110 -099	 .534 .440 .350 .283 .108 .058 .016 008 002 007 006 .020 .018		610 .512 .437 .371 .278 .191 .137 .092 .070 .071 .073 .076 .085		732 .627 .561 .493 .395 .309 .248 .205 .180 .184 .184 .188 .178	471 381 372 362 373 375 371 316 320 313 306 266 226 214	803 .657 .588 .521 .426 .342 .282 .239 .220 .213 .220 .223 .221 .223
0.75	1 .978 .955 .933 .912 .889 .867 .845 .800 .756 .711 .667 .622 .578	010034058078111137149156155150142138	075 .032020045065103107116111087		191 .133 .085 .053 .023 .005015022	- 284 - 279 - 294 - 305 - 331 - 331 - 323 - 308 - 285 - 274	287 .211 .171 .137 .109 .093 .072 .065 043 .041	- 375 - 384 - 386 - 386 - 406 - 411 - 413 - 412 - 405 - 387 - 387	.436 .369 .314 .279 .249 .231 .203 .191  .157 .152 .156 .153	- 489 - 476 - 478 - 475 - 474 - 472 - 469 - 469 - 446 - 421 - 414	.486 .410 .361 .326 .298 .280 .249 .239 .239 .205 .208 .205 .203						.410 .324 .262 .196 .153 .111 .086 .055 .059 .054	- 401 - 370 - 382 - 396 - 408 - 414 410 - 400 - 355 - 342	.527 .442 .383 .317 .267 .233 .217 .188 .193 .201 .189 .189	- 471 - 435 - 439 - 447 - 447 - 452 - 455 - 448 - 441 - 436 - 424 - 360 - 353	.558 .474 .419 .355 .310 .278 .270 .238 .241 .267 .226 .218 .199



TABLE IV.- CONTINUED

											= 5°										
		,				-				Left w	ing pane	1									
						e airfoi										Sharp-no					
Sta	tion	α=0	.1°	α=2.	5°	α=5.	20	α=8.	.6°	α=10	.1°	α=0.	.0°	α=2	.7°	α=5.	20	α=8.	3°	α=10	.1°
x/cr	x/2	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl
0.25	1 •933 •867 •800 •733 •667 •533 •400 •267	0.107 .048 .039 .041 .025 .017 .023	0.425 .204 .062 .040 .027 .026 .029 .025 .018	 -0.033 083 072 061  045 046 035 037	0.522 .317 .176 .145 .124 .112 .101 .088 .079	 -0.169 214 203 172 112 103 086 089	0.584 .423 .284 .247 .218 .197 .178 .158 .149	 -0.331 372 341 316  211 156 153	0.618 .541 .420 .379 .341 .315 .286 .261 .250 .241	 -0.365 396 375 353 256 202 180 186	0.647 .590 .486 .440 .398 .367 .332 .301 .287	 0.331 .250 .197 .165 .134 .095 .070 .061	0.331 .289 .233 .180 .158 .117 .070 .066 .046	0.170 .128 .092 .069 .046 .022 .006 .004	0.413 .360 .298 .244 .225 .180 .143 .121	 -0.012 .070 .022 .004 021 039 050 051 058	0.517 .453 .389 .332 .307 .259 .218 .194 .170	0.496419154127119124127122127	0.577 .534 .479 .427 .406 .359 .318 .293 .264	-0.549 486 368 156 161 150 157 138 145	0.627 .577 .519 .473 .454 .406 .366 .340
0.50	1 .967 .933 .900 .867 .800 .733 .667 .600 .533 .467 .407	054 014 036 048 060 071 068 063 066 056	.468 .069 014 400 039 480 065 072 071 065 060 060 048 053	117188175157148145148127124112103	.534 .229 .129 .091 .077 .044 .017 .003 .002 005 0	 257 312 320 308 279 245 221  197 185 163 149	.553 .350 .251 .206 .184 .140 .108 .087 .078 .070 .072 .068		.309 .258 .222 .199 .195 .175 .182 .172	510 506 490 472 448 432 277 269 274 252 234	.497 .516 .434 .388 .358 .304 .245 .240 .218 .218 .206 .191 .204	 .215 .145  .066 076 074 083 079 072 055 043	 .268 .197 .141 .094 .028 029 059 081 088 073 069 054 037	042 .008041 -098 -110 -122 -129 -139 -124 -103 -090 -089	361 .285 .224 .175 .107 .048 .015007011 .005 .011 .017				.528 .463 .415 .369 .299 .237 .199 .184 .176 .176 .182 .173 .168		575 .512 .464 .419 .350 .289 .253 .237 .229 .226 .231 .220 .212
0.75	1 .978 .955 .933 .912 .889 .867 .845 .800 .756 .622 .578 .538	113 127 134 131 125 116 108 097 092		239 244 242 231 229 220 207 193 180 165 159			  .192 .141 .108 .087 .067 .059 .048  .039 .048	528 529 520 520 505 491 473 453 423 389 295 275	.276 .244 .221 .200 .188 .170 .166	513 522 514 511 509 506 492 501 473 387 306		117 .064016063086126145141133127118106094	071 .0100310791101148161144131112104099	069071126159173214201194186173153153	  .176 .110 .066 .015 014 048 058 041 013  012		 .277 .210 .163 .113 .084 .060 .053 .041 .070 .085  .062		 .351 .325 .285 .239 .212 .197 .194 .192 .202 .192  .162 .157		 .439 .381 .342 .298 .271 .255 .250 .225 .241  .212 .211 .249

# TABLE IV. - CONTINUED

•										β =	90										
									R	ight wi		L									
				R	ound-nose	airfoi	1									Sharp-no	ose airf	oil			
St	ation	α==0	.10	α=2	.5°	<b>α</b> =5	.2°	α=8	3.6°	α=10	.1°	α=0.0	00	α=2	·7°	α=5.	.20	α=8.	.8°	α=10	.10
x/cr	у w/2	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl
0.25	1 .933 .867 .800 .733 .667 .533 .400 .267	0.193 .094 .049 .026 016 031 026 020	0.737 .382 .179 .112 .067 .032 .012 .007 004	0.117 .010 047 056  079 082 069 049	0.870 .536 .306 .235 .186 .150 .127 .101 .115				0.987 .745 .548 .473 .415 .370 .321 .275 .250			.456 .368 .291 .230 .176 .108 .063 .035	 .513 .454 .370 .291 .242 .166 .106 .059	 .317 .254 .187 .133 .085 .027 008 027 031	 .628 .543 .451 .366 .324 .244 .181 .130			252151119091080122140143128	.852 .755 .654 .569 .523 .436 .364 .312		
0.50	1 .967 .933 .900 .867 .800 .733 .667 .600 .533 .467 .400 .267 .133 0	.176 .041 021 039 060 099 112 111 113 100	.822 .265 .124 .074 .048 .008 029 059 069 073 081 082 		.930 .425 .280 .209 .184 .132 .085 .051 .031 .011 .011 .012 .024				.943 .691 .539 .458 .419 .340 .279 .239 .220 .191 .185 .177 			.385 .277 173 .068 .011 024 063 086 098 104 084 059	.482 .397 .316 .242 .154 .063 .006 075 078 061 039 031	.207 .143 057 027 077 104 136 156 164 168 141 112					.811 .694 .615 .537 .430 .335 .270 .218 .187 .176 .173 .183 .185 .175		
.75	1 .978 .955 .933 .912 .889 .867 .845 .800 .756 .711 .667 .622 .578 .533 .446				245 .167 .129 .092 .058 .037 .016 .005025025024028				483 .400 .345 .306 .271 .250 .220 .203163 .158 .164 .144			251 .193 .099 .039 .005055092121128139147144141	 .271 .188 .131 .070 .022 021 050 082 087 096 113 114 110					-341 -311 -327 -351 -367 -378 -380 -380 -379 -350 -349	 .587 .491 .426 .353 .297 .257 .257 .234 .201 .198 .190  .178 .183 .176		



TABLE IV.- CONCLUDED

										β = 9 <sup>C</sup>											
									Left	wing p	anel	,									
				Roun	nd-nose	airfoil									Sh	arp-nos	e airfo	oil			
St	ation	α=-0	0.10	α=2.	5°	α=5	.20	α=8	.6°	α=10	.10	α=0	.00	α=2	.70	α=	5.2°	α=8.	.8°	α=10	0.10
x/c <sub>r</sub>	y ₩/2	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl
0.25	1 •933 •867 •800 •733 •667 •533 •400 •267	 0.103 .049 .039 .046  .037 .027 .033 .021	0.347 .164 .048 .034 .029 .029 .031 .025 .019	 -0.039 076 063 046  033 036 026 037	0.438 .272 .153 .132 .115 .107 .100 .084 .078			 -0.367 384 339 323  191 173 150 160	0.516 .472 .384 .350 .318 .295 .273 .252 .244 .236			0.273 .211 .171 .144 .120 .089 .070 .065	 0.263 .238 .191 .149 .133 .100 .074 .058	 0.149 .105 .076 .059 .041 .022 .008 .008 016	 0.353 .307 .256 .215 .198 .162 .134 .117			 -0.441 409 261 178 123 112 121 133	0.523 .486 .437 .397 .382 .342 .310 .290 .261		
0.50	1 .967 .933 .900 .867 .800 .733 .667 .600 .533 .467 .400	 .054 007 	.362 .033 028 043 041 047 058 061 064 055 051 047 055		.434 .186 .112 .074 .064 .038 .017 .014 .015 002 .004 .004				.391 .431 .353 .309 .287 .244 .213 .197 .198 .177 .183 .177 .165 .157			173 .112047016038057069073069062051045	187 .137 .096 .058 .003042065080083071067091	018014044098104110112122111106094081094	-297 -230 -181 -142 -084 -036 -011 -002 -009 -001 -001 -009 -014 -002			595 -518 -490 -419 -336 -316 -279 -253 -246 -239 -227 -206 -191 -203	 .467 .412 .377 .340 .279 .230 .206 .193 .185 .189 .174 .174		
0.75	1 •978 •955 •933 •912 •889 •867 •845 •800 •756 •711 •667 •622 •578 •533 •446	062074093103115112105097091083078	 		049 .002 -013 -030 -041 -043 -046 -040036 -031 -025				 .312 .255 .230 .209 .191 .180 .165 .170  .163 .176			081 .041037075092125137124114108103093	014 -033 -067 -109 -133 -151 -157 -159 -137 -122108 -1055 -095		 .132 .073 .034 009 053 051 053 027 018 			 589 560 551  526 511  346 318 322 309 287 257	 -353 .284 .270 .232 .208 .198 .198 .173 .205 .188  .164 .167 .212		

TABLE V.— EXPERIMENTAL PRESSURE COEFFICIENTS [ M = 1.53 ]

	-										β =	00									
					Round-r	nose air	foil					]			\$	Sharp-no	ose airi	Coil			
St	ation	α	=-0.1°	α	=2.50	α.	=5.2°	α=	=8.6°	α=	=10.1°	α=	=0.0°	α=	2.7°	Œ.	=5.2°	α	=8.8°	α	=10.1
c/c <sub>r</sub>	¥ w/2	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pı
).25	.933 .867 .800 .733 .667 .533 .400 .267	0.177 .081 .034 .036  .021 .014 .012 .023	0.593 .257 .125 .080 .041 .035 .031 .028 .024	0 0 0 7 6 - 0 0 2 2 - 0 6 1 - 0 6 0 0 5 2 - 0 4 9 - 0 4 6 - 0 3 1	0.671 .360 .222 .173 .114 .110 .100 .089 .079	 -0.032 -117 -155 -152  -118 -103 -093 -076	0.722 .458 .317 .266 .215 .206 .179 .159 .146 .140	- 0 162 - 216 - 241 - 252 - 220 - 184 - 159 - 133	0.767 .565 .453 .405 .344 .325 .286 .258 .238	 -0.211 -259 -272 -286  -247 -211 -173 -151	0.757 .605 .483 .434 .376 .354 .314 .284 .264	0.364 .291 .225 .177 .146 .106 .077 .064	0.380 327 272 222 185 134 094	0.242 .201 .148 .105 .077 .041 .019 .011	0.486 .415 .351 .297 .258 .201 .156 .129	0.064 .095 .052 .023 .006 011 020 023	0.563 .486 .421 .368 .334 .276 .222 .193	-0.286 -191 -152 -090 -088 -111 -115 -105 -104	0.667 .596 .530 .477 .440 .376 .322 .291	 -0.377 -279 -242 -204 -153 -139 -137 -125 -125	0.68 .62 .55 .5X .41 .41 .35 .33
0.50	1 967 933 900 867 800 733 667 600 533 467 400 267 133		.589 .192 .090 .032 .012 011 032 046 051 056 046 049 044 037	016 -081110 -130 -133 -142 -142 -133 -125 -123 -109 -100 -096	.620 .293 .190 .140 .118 .084 .025 .015 .009 .015 .007 .012	104 - 175 217 - 227 - 229 - 213 - 193 - 187 - 173 - 155 - 143 - 138	.649 .418 .303 .246 .219 .171 .127 .100 .087 .083 .070 .067 .070	- 260 - 289 - 297 - 308 - 310 - 324 - 307 - 292 - 275 - 243 - 205 - 192 - 186	.650 .552 .440 .385 .349 .288 .240 .210 .193 .187 .174 .160 .159	328 - 339 - 329 - 334 - 351 - 292 - 309 - 302 - 270 - 215 - 200 - 202	.630 .580 .470 .407 .377 .317 .268 .238 .224 .216 .206 .199 .185 .181	286 204 162 129 056 015 -017 -042 -051 -062 -059 -042 -027	-341 -271 -213 -165 -098 -035 -051 -053 -051 -031 -016 -010	- 128 .085 .053 .026 - 032 064 088 108 124 118 098 082 069	.458 .367 .299 .250 .172 .108 .065 .031 .015 .013 .017 .032 .046	181 092 026 036 113 144 158 173 177 169 147 126	.545 .454 .380 .329 .251 .184 .134 .096 .072 .074 .073 .087	352 282 257 253 259 259 230 238 238 238 24 168 155	-641 .552 .486 .437 .358 .292 .245 .208 .185 .178 .177 .186	- 432 - 351 - 324 - 325 - 325 - 325 - 314 - 257 - 258 - 255 - 243 - 2190 - 178	. 67 . 58 . 52 . 46 . 38 . 31 . 26 . 23 . 21 . 21 . 21 . 21
0.75	978 975 933 912 889 867 845 800 756 711 667 622 578 •333	-008 -021 -046 -063 -091 -107 -120 -115 -115 -116 -103			.163 .118 .048 .022 .009 006 011 028 031 024	- 228 - 231 - 244 - 254 - 272 - 281 - 285 - 278 - 263 - 278 - 238 - 230	.264 .208 .138 .107 .092 .072 .066 .046 .044 .044 .049	- 334 - 342 - 345 - 347 - 358 - 361 - 363 - 359 - 342 - 332		- 350 - 369 - 366 - 369 - 375 - 383 - 384 - 387 - 382 - 364 - 349	424 .371 293 .266 .250 .227 .214 .199 188 .179 .188 .179	-204 -162 -087 -046 -010 -069 097 -104 -107 -108 -104		- 045 030 - 018 - 048 - 080 - 124 - 164 - 166 - 169 - 166 - 160 - 146	- 285 218 .166 .112 .076 .047 -011 -008 -012 014 -010	212 193 202 199 215 195 217 235 234 231 226 212 197	361 288 230 176 139 109 075 047 048 052	- 355 - 332 - 319 - 326 - 341 336 - 341 295 - 289 - 289 - 264	.461 .395 .337 .285 .251 .227 .204 .179 .184 .182	- 427 - 396 - 396 - 398 - 401 - 384 - 370 - 330 - 298 - 284	



TABLE V. - CONTINUED

							The state of			β =											
-					Round-n	ose air	foil			Right w	ing pane	1			9	harp-no	ge simf	017			
Sta	tion	α=	-0.1°	α=	2.50	α=	5.2°	α=	8.6°	α=	10.10	α=	0.00	α=	2.7°		5.20		8.8°	α=	10.10
x/c <sub>r</sub>	<u>n/2</u>	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pı	Pu	Pı	Pu	Pı	Pu	Pl	Pu	Pl
0.25	.933 .867 .800 .733 .667 .533 .400 .267	0.230 .111 .049 .042  .011 0. 003	0.715 .329 .165 .108 .055 .041 .023 .020 .012	 0.131 .021 035 043  068 068 063 047	0.777 .429 .268 .207 .133 .120 .101 .087 .073	 0.019 075 124 130  145 128 113 094	0.832 .519 .360 .297 .231 .215 .181 .157 .137	 -0.097 165 205 217  236 206 175 142	0.897 .657 .494 .434 .363 .339 .293 .259 .234	 -0.164 234 263 270  261 225 188 158	0.877 .677 .518 .458 .395 .368 .320 .285 .259 .241	0.418 .339 .262 .204 .166 .112 .076 .056 .039	0.450 .388 .323 .261 .216 .152 .101 .067	0.287 .238 .173 .121 .088 .041 .009 004	0.561 .481 .407 .343 .296 .226 .166 .127	0.083 .131 .078 .036 .010 031 050 055 060	0.646 .555 .478 .412 .366 .295 .230 .189 .169	-0.193 119 089 074 103 111 104 099	0.746 .660 .579 .515 .470 .397 .333 .288 .263	 -0.277 198 169 158 144 134 142 131 125	 0.786 .693 .619 .553 .510 .436 .370 .325 .296
0.50	1 .967 .933 .900 .867 .800 .733 .667 .600 .533 .467 .400 .267 .133	196 .071019018039070078082081081060055	.710 .251 .129 .068 .042 .007 028 051 060 064 065 058 048	081037	.755 .363 .237 .178 .147 .104 .061 .031 .016 .004 001 003 .003		.773 .469 .334 .281 .247 .191 .139 .106 .087 .074 .053 .042 .043		.805 .631 .496 .386 .317 .258 .219 .197 .176 .160 .153 .149	- 269 - 299 - 313 - 323 - 326 - 341 - 312 - 322 - 308 - 291 - 206 - 203	.775 .650 .523 .447 .405 .380 .242 .220 .205 .195 .184 .174 .175 .167	354 -262 -213 -175 -088 -037 -031 -050 -061 -065 -057 -043 -033	433 -349 -279 -222 -149 -076 -025 -013 -037 -047 -051 -043 -015 -019	204 .151 .109 .075 .002045103117128128126082068	 .563 .462 .379 .312 .228 .145 .688 .045 .028 .008 .002 .013 .025 .033		619 .512 .433 .371 .282 .203 .147 .101 .075 .068 .066 .076 .085		.718 .619 .538 .477 .386 .246 .199 .166 .160 .157 .164 .166		.765 .661 .576 .514 .423 .344 .282 .237 .204 .193 .201 .200 .196
0.75	.978 .975 .933 .912 .889 .867 .711 .667 .622 .578 .533	 .037 .003 026 044  078 102 118 124 123 129 120	 .096 .023  021 046 060 080 087 104  105 107 103 095		.210 .158  .079 .049 .032 .012 .001 022  027 037 035 034		.305 .247 .164 .127 .110 .084 .076 .060 .043 .035 .050 .044				 .458 .396  .311 .280 .262 .234 .220 .201  .178 .176 .181	255 .206 .121 .073 .032025056077108117123121	 .253 .182 .123 .072 .034 .002 033 066 077 087 	.095 .080 .020 014 049 101 130 159 160 168 174 178		166150158166185175198226232234224220	.409 .326 .273 .213 .170 .139 .070 .066 .056		.522 .438 .372 .314 .273 .241 .210 .177 .174 .168 .158	- 369 - 344 - 342 - 355 - 367 364 - 360 - 356 - 348 - 321 - 308	 .5599 .475 .4099 .3511 .280 .250 .217 .220 .222  .202 .198



TABLE V.- CONTINUED

										β = 5										- 276	
									Lef	t wing	panel										
			0.10		d-nose		5.20		8.60	,	0.10	α=0	00	1		rp-nose		1	00		- 0
Stat			1	-	2.5	α=	).2		1	α=1	0.1		1	α=2	.70	α=5	1.20	α=8	1	α=1	0.10
x/c <sub>r</sub>	<u>y</u> w/2	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pı	Pu	Pı
0.25	1 •933 .867 .800 .733 .667 •533 .400 .267	0.145 .064 .039 .043 049 .022 .022	0.407 .187 .074 .049 .019 .025 .025 .024 .019	0.026 043 059 053 	0.546 .286 .181 .146 .101 .100 .091 .081 .073	 -0.088 147 159 145  097 087 081 076	0.595 .387 .282 .244 .202 .196 .173 .157 .146 .138	-0.243 290 285 269 189 156 141 135	0.632 .507 .412 .373 .327 .308 .278 .254 .239	 -0.262 281 293 293  242 187 166 150	0.667 .555 .469 .429 .375 .356 .321 .294 .279 .269	0.306 .245 .192 .156 .133 .103 .079 .076	0.288 .258 .215 .177 .151 .111 .082 .076	0.195 .157 .115 .086 .070 .052 .037 .034	0.406 .348 .297 .257 .229 .181 .144 .126	 -0.025 .075 .034 .011 002 014 026 027 034	0.506 .438 .383 .339 .310 .259 .218 .196	 -0.397 274 190 092 097 101 104 094 106	0.577 .523 .473 .431 .404 .351 .310 .287	 -0.436 330 279 179 115 115 112 098 117	0.522 .571 .521 .480 .453 .401 .357 .332
0.50	1 .967 .933 .900 .867 .800 .733 .667 .600 .533 .467 .400	095 .012 009 029 037 052 051 050 048 049 042 038 041	.428 .101 .012 015 022 029 045 055 056 055 052 044 040 046	049 - 121 125 - 127 - 123 - 113 - 116 - 119 - 113 - 110 - 099 - 091 - 093	.489 .223 .135 .106 .088 .058 .011 .006 .008 .004 .007 .004	.169 .229 234 215 196 187 170 151 136 128 130	.523 .356 .263 .215 .192 .150 .115 .086 .086 .077 .075 .072	- 368 - 376 - 368 - 356 - 342 - 319 - 282 - 243 - 208 - 208 - 186 - 183 - 182	.514 .481 .389 .343 .313 .262 .222 .198 .191 .177 .174 .172 .166 .160		.509 .531 .442 .399 .366 .312 .269 .249 .237 .222 .220 .209 .197 .197	- 216 .151 .115 .090 .029 004 024 042 050 051 045 031 018	- 238 .181 .136 .099 .045 004 033 053 062 060 053 032 023	- 086 .052 .025 .038 - 038 - 066 - 088 - 102 - 109 - 110 - 102 084 - 071	.368 .295 .233 .191 .123 .074 .040 .018 .004 .005 .011 .028		- 1 - 472 - 393 - 327 - 284 - 215 - 161 - 120 - 091 - 081 - 085 - 089 - 108 - 101 - 094		.548 .478 .417 .377 .310 .255 .219 .197 .186 .189 .177	475 410 382 369 363 348 256 242 240 237 224 199 182 178	586 .514 .464 .424 .355 .301 .262 .239 .232 .233 .235 .234 .219
0.75	1 .978 .955 .933 .912 .889 .867 .845 .800 .756 .711 .667 .622 .578 .533 .446			139 164 179 187 199 199 191 170 171 150 153		- 257 - 265 - 271 - 278 - 291 - 295 - 284 - 246 - 229 - 206 - 200	.208 .175 .108 .083 .075 .060 .061 .073 .052 .050 .062			393 - 401 - 401 - 401 - 405 - 405 - 405 - 393 - 388 - 379 357		147 .106 .042 .006 025 	-106 .054 .011 -039 -053 -071 -092 -107 -100 -097 088 -079	008 006 055 076 107 142 158 153 147 137 123		- 260 - 238 - 236 - 222 - 225 	.306 .241 .191 .144 .114 .092 .070 .053 .070 .077		-393 .334 .288 .247 .222 .205 .194 .175 .199 .185		 .435 .381 .336 .298 .275 .260 .250 .228 .245 .230  .213 .204



# TABLE V.- CONTINUED

										β =	9°										
				Davis	nose ai	0-17			Ri	ight win	g panel										
04-	+1.		0		-			T	- 10			-					airfoi	1			
Sta	tion	α=	0.10	α=2	2.50	α=5	.2°	α=8	3.60	α=]	0.10	α=0	0.00	α=2	.70	α=	5.20	α=8	.80	α=	10.10
x/cr	<u>y</u> w/2	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl
0.25	1 .933 .867 .800 .733 .667 .533 .400 .267	0.267 .144 .070 .056  005 024 028 018	0.776 .376 .207 .134 .072 .047 .013 0 008	0.196 .052 017 026  081 087 083 062	0.895 .510 .309 .237 .153 .124 .100 .079 .062			- 0.043 - 129 - 181 - 192 229 - 230 - 197 - 154	0.997 .707 .526 .452 .373 .344 .290 .251 .218			0.485 .387 .299 .233 .188 .121 .074 .047	0.526 .447 .371 .297 .246 .170 .105 .062	0.321 .270 .199 .139 .098 .037 0 019 030	0.621 .529 .447 .371 .316 .234 .165 .118			 -0.104 046 036 035 046 081 109 115 113	0.841 ·734 ·638 ·562 ·510 ·424 ·346 ·290 ·254		
0.50	1 .967 .933 .900 .867 .800 .733 .667 .600 .533 .467 .400 .267	245 .103 - 046 .001 - 026 - 067 - 072 - 093 - 097 - 100 - 090 - 076 068	.818 .306 .170 .110 .076 .032 012 043 060 069 074 072 061	-137 .013 -050 083 107 138 161 167 163 159 139 123 109	.887 .436 .290 .215 .181 .080 .044 .025 .008 0 005 001			099 188 2 - 1 249 262 282 297 301 303 291 249 271 192	.928 .681 .529 .451 .408 .332 .267 .223 .196 .173 .164 .155 .146			437		- 251 .190 .144 .107 .021 057 087 107 123 125 127 103 087	.611 .500 .416 .347 .257 .170 .107 .058 .022 .006 002			- 202 147 157 160 181 207 221 230 246 254 254 253 253 172	.827 .709 .594 .426 .331 .273 .220 .183 .169 .163 .176		
0.75	1 .978 .975 .933 .912 .889 .867 .845 .800 .756 .711 .667 .622 .578		 .146 .103 		 -257 .180  .109 .077 .035 .017 -011  .024 035 038				 .478 .411  .307 .272 .250 .219 .200 .179  .147 .136					.121 .102 .050 .013 023 083 120 153 163 172 176				- 229 - 217 - 229 - 254 273 - 282 - 294 - 291 - 292 - 290 - 282	.607 .502 .430 .365 .331 .282 .243 .208 .200 .189		

TABLE V.- CONCLUDED

										β =	/										
					Round-	nose ai	rfoil		I	eft win	ng panel	-				Shown-no	se airf	oil			
Stat	ion	~	=-0.1°	~-	2.5°		5.2°		-8.6°		=10.1°		:0.0°		2.70	_			0.00		
Dual			1	u-	-•>	- U-	).2	a-	-0.0	Q.=	10.1	α=	1	α=	2.7	α=	5.2°	α=	8.8°	α=	10.1
/c <sub>r</sub>	w/2 y	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pı
25	.933 .867 .800 .733 .667 .533 .400 .267	0.119 .066 .044 .052  .041 .035 .033 .027	0.367 .146 .060 .043 .024 .033 .034 .031 .027	0.017 046 054 040 	0.479 .262 .165 .136 .099 .101 .093 .083 .077			 -0.275 310 279 256  162 139 129 133	0.536 .470 .382 .352 .311 .298 .271 .253 .242 .232			0.252 .225 .181 .153 .138 .117 .102 .102 .066	0.285 .222 .190 .161 .143 .114 .091 .081	 0.159 .130 .099 .076 .064 .048 .034 .033	 0.353 .307 .268 .237 .214 .177 .147 .131			 -0.430 281 176 106 080 078 081 071	 0.522 .477 .438 .405 .383 .340 .309 .292 .268		
).50	.967 .933 .900 .867 .800 .733 .667 .600 .533 .467 .400 .267 .133		.340 .063 0 019 020 026 039 045 041 033 037 034 042		.430 .201 .123 .097 .083 .056 .030 .016 .012 .015 .011			414 - 409 - 381 - 351 - 337 - 322 - 253 - 209 - 194 - 195 - 178 - 179	.419 .444 .366 .326 .300 .256 .233 .208 .195 .181 .184 .179 .164 .163 .158			 .210 .150 .113 .090 .036 .008 012 033 053 053 053 053 023 017 018	202 .160 .116 .083 .036005029047055094048024027	 .066 .037 .015 .001 044 061 073 082 086 083 077 064 067	.315 .252 .202 .165 .106 .062 .034 .009 0 .003 .007 .023 .028			- 483 - 422 - 377 - 363 - 317 - 205 - 200 - 206 - 201 - 196 - 183 - 167 - 156 - 158	 .473 .414 .375 .341 .285 .211 .193 .186 .185 .185 .185 .185		
0.75	1 .978 .955 .933 .912 .889 .867 .845 .800 .756 .711 .667 .622 .578 .578 .446			- 158 - 180 - 188 - 189 - 192 - 189 - 184 - 172 - 153 - 134 - 135	 .081 .056 003 020 025 030 026 036 			451 - 453 - 449 - 449 - 445 - 426 - 418 - 412 - 362 - 295 234	.297 .258 .201 .184 .174 .163 .161 .153 .150 .148			- 116 .085 .021 - 009 034 068 079 061 075 069 057 057	054 .009 026 069 079 079 107 115 100 090 074 064						 .344 .298 .260 .227 .209 .199 .190 .172 .192 .178		

TABLE VI.— EXPERIMENTAL PRESSURE COEFFICIENTS [M = 1.60]

										β = (	00					Name of the last					
			1		Round-	nose ai	rfoil								Sh	arp-nos	se airfo	il			
Stati	ion	α=-	-0.1°	α=2.	5°	α=5.	20	α=8.	6°	α=10.	10	α=0	0.0°	α=2	2.7°	α=5	.2°	α=8	3.8°	α=10	).1°
x/c <sub>r</sub>	<u>y</u>	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	P2	Pu	Pl	Pu	Pl	Pu	Pl
0.25	1 .933 .867 .800 .733 .667 .533 .400 .267	.158 .077 .042 .030 .009 0	0.525 .269 .112 .067 .038 .019 .014 .013 .007	-070 -014 -049 -056 052 -053 -039 -038	0.626 .384 .207 .158 .124 .102 .087 .074 .066	 057 111 125 146  137 120 097 101	0.646 .428 .275 .224 .183 .154 .130 .115 .100		0.744 .577 .420 .367 .325 .294 .262 .229 .213 .209	 192 237 246 265  239 207 158 152	0.747 .608 .469 .416 .371 .338 .300 .266 .249	.345 .261 .203 .159 .122 .077 .053 .040 .030	.352 .305 .243 .185 .156 .105 .071 .046 .036	.219 .174 .126 .087 .054 .016 004 010	.461 .400 .328 .266 .232 .175 .133 .102 .087	.026 .060 .036 .001 024 043 064 062 071	.522 .458 .384 .321 .287 .227 .181 .149 .129	 267 186 159 128 119 135 136 127 131	.641 .556 .478 .412 .382 .315 .263 .228 .206	 	.676 .619 .545 .481 .449 .389 .331 .296 .271
0.50	1 .967 .933 .900 .867 .800 .733 .667 .600 .533 .467 .400 .267 .133	161 .052013017036077077077079058	.646 .198 .091 .045 .024 -005 -033 054 062 069 069 068 052 050	 .047 053  088 111 126 139 137 131 127 113	.717 .325 .200 .147 .126 .086 .048 .019 .004 008 008		.599 .381. .266 .214 .184 .131. .087 .057 .042 .039 .028 .023		.731 .569 .454 .385 .355 .296 .244 .208 .201 .165 .159 .159	281 - 298 - 296 - 296 - 308 - 306 - 326 292 - 289 - 279 - 218 - 193	.710 .601 .492 .432 .398 .330 .273 .234 .222 .195 .188 .184		368 .296 .229 .173 .105 .031 - 044 - 068 - 067 - 077 - 071 - 051 - 043	.160 .110 .042 028 064 091 116 129 134 113 113 955 086	477 .392 .325 .262 .179 .103 .016007013011002		.529 .440 .371 .307 .220 .140 .096 .067 .042 .047 .049 .059		 .646 .561 .502 .441 .354 .276 .224 .186 .159 .149 .146 .134 .134		.679 .600 .545 .486 .396 .315 .255 .215 .194 .192 .189 .181
0.75	1 .978 .955 .933 .912 .889 .867 .845 .800 .756 .711 .667 .578 .533 .446		067 .030010053063077089088084070		180 .119 .086 .059 .031 .016 .001004019020002		245 .207 .146 .119 .091 .078 .056 .049036 .040 .052		394 ·317 ·283 ·251 ·208 ·187 ·177 		 .449 .385 .336 .307 .275 .258 .233 .222  .197 .196 .193 .183	216 .166 .090 .042 .018032 -058077 -082 -091 -096								383 - 353 - 348 347 357 - 357 - 358 - 357 - 358 - 357 - 358 - 357 - 358 - 379 - 293 - 279	 .512 .442 .388 .330 .288 .258 .242 .242 .247 .236  .209 .204

# TABLE VI.- CONTINUED

										β =	50										
										Right wi	ng pane	1								V. T. W.	
					Round-n	ose airf	oil								2	harp-no	se airf	oil	The T		
S.	tation	α=	-0.1°	α=	2.5°	α=	5.2°	α=	8.6°	α=	10.1°	α=	0.0°	α=	2.7°	α=	5.2°	α=	8.8°	C=	10.10
x/c <sub>r</sub>	y w/2	P <sub>u</sub>	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl
0.25	1 .933 .867 .800 .733 .667 .533 .400 .267	0.202 .107 .056 .036  003 016 007 006	0.641 .347 .157 .096 .057 .025 .005 .003 004	0.110 004 028 045 076 077 060 053	0.732 .445 .249 .184 .141 .105 .079 .064 .052	 -0.002 077 113 129  149 133 105 100	0.767 .513 .327 .260 .212 .177 .147 .126 .109 .105	-0.088 -165 -198 -209  -226 -200 -159 -143	0.866 .653 .464 .397 .345 .305 .261 .226 .204 .194	-0.150 -209 -236 -246 -253 -222 -176 -160	0,860 .684 .507 .439 .387 .344 .300 .263 .239 .226	0.391 .304 .236 .181 .134 .076 .044 .025	0.410 .359 .286 .218 .177 .112 .068 .034	0.265 .211 .152 .103 .062 .010 017 029 033	0.527 .456 .373 .298 .256 .184 .132 .093	0.105 .097 .063 .024 008 050 068 066	0.595 .517 .431 .357 .315 .243 .190 .149 .129		0.723 .650 .561 .483 .440 .360 .295 .249		0.753 .682 .595 .519 .478 .398 .333 .286
0.50	1 .967 .933 .900 .867 .800 .733 .667 .600 .533 .467 .400 .267 .133 0	- 185 .063 - 013 - 025 - 048 - 079 - 092 - 091 - 092 - 092 - 092 - 096 - 082 - 069 .008	.726 .233 .112 .059 .034 004 037 075 075 075 075 056	- 075 - 022 - 079 - 109 - 128 - 154 - 159 - 152 - 148 - 130 - 126	.788 .339 .214 .155 .127 .082 .041 .013 002 014 017 019 017 08	-044 -133 -161 -170 -191 -203 -224 	.689 .431 .302 .245 .215 .160 .114 .081 .064 .051 .044 .039	158213251265273290277275260223199	.829 .596 .462 .392 .358 .289 .232 .192 .176 .152 .146	- 234 - 265 - 283 - 286 - 300 - 304 - 321 - 310 - 298 - 285 - 240 - 203	.805 .631 .505 .433 .396 .230 .212 .189 .185 .175 .175	333	398 ·322 ·251 ·187 ·112 ·038 - 009 ·044 - 071 - 076 - 081 - 069 - 051 - 045	.180 .124 .049 -028 -065 -097 -120 -132 -151 -132 -111	514 .417 .343 .274 .187 .106 .057 .014 016 020 024 010		.566 .476 .402 .339 .252 .168 .077 .050 .053 .045 .045	- 259 - 203 - 200 - 202 - 224 - 236 - 247 - 248 - 258 - 261 - 255 - 198 - 185	.699 .605 .527 .464 .366 .276 .220 .174 .145 .142 .133 .143	311 - 248 - 246 - 243 - 263 - 272 - 281 - 283 - 279 - 270 - 240 - 213 - 198	
0.75	1 .978 .955 .933 .912 .889 .867 .845 .800 .776 .711 .667 .628 .578 .533 .446	.072 .038 .007 -016 -050 -076 094 105 111 115 117			-236 .171 .131 .100 .067 .048 .024 .010 	- 149 - 162 - 181 - 195 - 216 - 230 - 238 - 249 - 256 - 253 - 244 - 244		- 273 - 283 - 286 - 286 - 286 - 317 - 314 - 317 - 313 - 304 - 307	.454 .379 .330 .294 .261 .215 .203 .163 .159 .144	- 314 - 332 - 328 - 335 - 346 - 351 - 352 - 346 - 333 - 331	.496 .420 .372 .335 .304 .284 .253 .240 .203 .194 .185	- 268 -218 -133 -078 -050 010 044 	- 266 .196 .144 .092 .012 -017 -048 -055 -067 089 -100	-119 104 043 -002 -026 -117 -146 -160 -169 -174 -185				- 289 - 264 - 273 287 - 300 - 306 311 - 312 - 307 - 302 - 277 - 287	.544 .464 .463 .336 .286 .251 .224 .183 .198 .198	324 - 243 - 305 316 - 331 332 - 332 - 333 - 330 - 309 - 304	584 504 504 376 326 289 271 226 250 237 - 205 193



TABLE VI.- CONTINUED

										β = 5°									-		
									Left	wing p	anel										
					Round-no	se airfo	oil						WHEN T			Sharp-n	ose air	foil		-	MIN
Sta	ation	α=-	0.10	α=2.	50	α=5.	.20	α=8.	60	α=10	.10	α=0.	00	α=2.	70	α=5.	20	α=8.	80	α=10	0.10
x/c <sub>r</sub>	<u>y</u> w/2	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl
0.25	1 .933 .867 .800 .733 .667 .533 .400 .267	0 .125 .059 .038 .032  .018 .009 .018	0.412 .202 .077 .043 .023 .012 .013 .013 .006	0 •018 041 055 057 045 046 031 037	0.494 .298 .165 .127 .101 .086 .078 .067 .062	0 066 116 133 134  103 095 076 079	0.576 .398 .267 .226 .195 .174 .155 .137 .126	0 193 227 233 243 193 162 130 132	0.618 .508 .388 .346 .308 .280 .251 .227 .214 .206	0 217 247 260 268 228 185 150 149	0.653 .566 .449 .402 .361 .332 .299 .271 .254 .245	0 .305 .230 .180 .145 .115 .079 .060 .053 .035	0 .286 .253 .202 .156 .133 .092 .067 .047	0 .177 .134 .095 .066 .043 .017 .005 .002	0 .391 .335 .273 .223 .198 .154 .123 .102	0 015 .058 .035 .012 016 039 046 043 053	0 .486 .421 .354 .301 .273 .224 .188 .163 .143	0 322 231 174 080 094 112 113 104 117	0 .577 .522 .459 .407 .379 .324 .285 .260 .235	0 362 277 236 184 118 127 126 118 134	.617 .579 .514 .459 .432 .376 .333 .305 .278
0.50	1 .967 .933 .900 .867 .800 .733 .667 .600 .533 .467 .400 .267	116 .029004019033049057061059	.520 .129 .037 .005 008 019 040 052 055 062 061 054 056		.561 .245 .152 .113 .098 .069 .040 .021 .012 .001 .001 003 004	 -114 -179 -190 -192 -199 -191 -179  -166 -153 -147 -133	.595 .361 .266 .224 .202 .161 .125 .105 .094 .083 .087 .082		.570 .490 .392 .349 .322 .270 .230 .207 .204 .178 .179 .168		.558 .531 .442 .395 .369 .313 .274 .251 .250 .221 .219 .219	259 .186 .215 .107 .039 .009 -015035049054055046037040	- 291 - 230 - 174 - 129 - 068 - 012 - 044 - 058 - 057 - 057 - 045 - 045 - 042 - 055	103 .063 014 038 060 078 095 105 109 105 094 083 083	399 318 .277 .215 .146 .085 .052 .020 .009 .012 .004 .006 .007 003	 -169 -090 -016 -034 -098 -117 -128 -139 -151 -145 -141 -130 -117	.490 .412 .351 .302 .228 .164 .127 .104 .086 .089 .080		563 .494 .447 .398 .324 .262 .221 .202 .187 .178 .169 .149	391 - 341 - 335 - 326 - 328 - 319 - 276 - 238 - 243 - 219 - 173 - 178	.599 .532 .484 .435 .362 .302 .245 .233 .229 .230 .224 .217
0.75	1 .978 .955 .933 .912 .889 .867 .845 .667 .662 .533 .446				 .118 .086 .044 .022 .001 007 015 013  023 020 017 009					348 - 365 - 357 - 359 - 362 - 369 - 361 - 356 - 346 - 334 - 339	405 .367 .311 .285 .257 .243 .221 .216 194 .194 .192 .184	 .153 .114 .034 008 031  070 090  095 091 088 086 066				226 -202 -202 -214 -199 	.317 .249 .204 .155 .124 .095 .079 .065 .079	- 386 - 358 - 358 - 357 - 357 - 362 - 360 349 - 300 - 289 - 287 - 266			

## TABLE VI.- CONTINUED

										β =											
									R	ight wi	ng panel			W.						_	
				RC	ound-nos	e airfo	oil								Sha	rp-nose	e airfo	il			
Sta	ation	α=-0	).1°	α=2.	.50	α=5.	20	α=8.	.6°	α=10	).1°	α=0.	.00	α=2.	.70	α=5	.20	<b>α</b> =8	.80	α=1	0.10
x/c <sub>r</sub>	<u>у</u> w/2	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl
0.25	.933 .867 .800 .733 .667 .533 .400 .267	0.253 .141 .079 .055 	0.776 .422 .205 .133 .087 .045 .012 .001 006	0.145 .051 004 024 	0.817 .496 .287 .214 .163 .119 .084 .066 .050			 -0.047 131 174 185  229 216 172 150	0.950 .708 .408 .414 .356 .308 .263 .223 .196 .181			0.418 .330 .255 .193 .143 .073 .034 .011	0.448 .392 .315 .239 .193 .118 .067 .026	0.291 .230 .168 .112 .066 0 031 048	0.577 .493 .402 .319 .269 .188 .132 .084			 -0.109 055 054 063 082 124 144 141	0.780 .701 .605 .511 .472 .380 .306 .250		
0.50	1 .967 .933 .900 .867 .800 .733 .667 .600 .533 .467 .400 .267 .133	-238 .104 -038 .006 -036 -073 -106 -109 -112 -102 -087	.867 .308 .169 .103 .073 .028 014 047 066 079 083 083	-120 0 052 052 111 141 168 169 169 149	.905 .398 .262 .190 .159 .107 .060 .023 .003 013 022 029 016			- 100 - 178 - 226 - 242 - 255 - 274 - 286 - 287 - 277 - 243 - 210	.919 .639 .492 .416 .379 .306 .244 .200 .177 .150 .139 .131			- 360 .262 - 161 .064 .017 021 057 083 101 112 106 085 072	441 .344 .283 .215 .136 .055 .003 038 071 081 085 088 068	.212 .152 .064 018 061 100 125 149 164 174 162 136	.571 .469 .378 .304 .211 .063 .012 020 033 045 019			199 153 153 160 191 228 241 259 265 271 249 217 197	.761 .658 .574 .507 .403 .305 .239 .186 .149 .141 .121		
0.75	1 .978 .955 .933 .912 .889 .867 .845 .800 .756 .711 .667 .622 .573 .446	103 .062 .024 003 079 101 115 124 132 137	 .177 .108 .070 .034 .002 022 046 064  103 110					- 239 - 236 - 249 - 253 - 269 - 281 - 298 - 300 - 302 - 298 - 304	.500 .416 .364 .321 .275 .255 .201 .138 .125 .126			 .288 .235 .145 .086 .053  012 050  089 106 119 128 128	.298 .221 .164 .107 .060 .015 017 055 065 077 102 111	-122 .104 -018 -043 -037 -100 -132 161 -177 -199 -189 -207	.407 .312 .247 .180 .127 .080 .044 .003 .010 024 053			- 222 - 196 - 209 240 - 264 - 277 290 - 291 - 295 - 296 - 282 - 300	.609 .526 .459 .385 .325 .284 .247 .201 .187 .166		

TABLE VI .- CONCLUDED

										β =											
				D 3		0.17				ert wir	ng panel		-								
			0		nose ai										Sharp	nose a	irfoil				
	tion v		-0.1°	α≈	2.5°		5.2°	α=	8.6°	α=	10.1°	α=	0.00	α=	2.70	α=	5.20	α=	8.80	α=	10.10
x/c <sub>r</sub>	₩/2	Pu	Pı	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pı	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pı	Pu	P7
0.25	.933 .867 .800 .733 .667 .533 .400 .267	0.130 .077 .062 .062 040 .029 .034 .018	0.375 .193 .080 .056 .043 .036 .033 .027 .019	 0.017 030 031 022  020 024 014 032	0.459 .283 .170 .141 .119 .106 .093 .080 .071			 -0.211 247 234 221  150 132 111 130	0.581 .517 .400 .363 .329 .302 .274 .255 .230 .215			 0.335 .243 .196 .165 .135 .098 .071 .062 .034	0.336 .263 .214 .165 .150 .108 .073 .052	0.179 .140 .107 .085 .063 .037 .021 .017	0.375 .331 .273 .227 .205 .160 .129 .109			 -0.354 286 100 048 068 076 085 083 113	 0.556 .511 .452 .408 .386 .335 .295 .271		
0.50	1 .967 .933 .900 .867 .800 .733 .667 .600 .533 .467 .400 .267 .133	.090 .024 .006 012 019 027 029 024 025 021 023	.390 .073 .005 011 013 023 032 039 039 033 029 027 040		.458 .199 .126 .098 .087 .062 .039 .026 .023 .019 .017			345 - 348 - 335 - 329 - 319 - 302 - 291 205 - 182 - 162 - 152	.443 .440 .361 .318 .295 .252 .200 .205 .186 .189 .187				259 .203 .133 .093 .040 010 032 049 054 049 039 025 019 042	070 .042003036050063085081073058049069	 .329 .263 .242 .200 .141 .072 .045 .036 .029 .034 .041 .039				.481 .426 .389 .348 .286 .237 .209 .194 .187 .185 .189 .191		
0.75	1 .978 .955 .933 .912 .889 .867 .845 .800 .756 .711 .667 .622 .578 .533 .446				.070 .045 .005 .005 031 027 032 036 029 023 010				.301 .269 .224 .205 .187 .179 .163 .170  .157 .157			123 .068 -012 -048 -071104 -119115 -106 -098 -089 -074	 .054 008 049 127 135 127 125 117 085 085		149 .103 .065 .023002024034046027014003				338 .286 .253 .214 .191 .179 .165 .198 .190 -160 .154		

## TABLE VII.— EXPERIMENTAL PRESSURE COEFFICIENTS [M = 1.70]

		1								β =	00										
					nose a				-							nose air			0.00		0
Sta	tion	α=	-0.1°	α=	2.5°	α=	5.2°	α=	8.6°	α=:	10.1°	α <sub>F</sub>	0.0°	α=	2.70	α=	5.20	α=	8.80	α=:	10.10
c/c <sub>r</sub>	<u>y</u>	Pu	Pı	Pu	Pı	Pu	Pl	Pu	Pl	Pu	Pı	Pu	Pı	Pu	Pı	Pu	Pı	Pu	Pı	Pu	Pı
0.25	1 .933 .867 .800 .733 .667 .533 .400 .267	0.210 .112 .061 .050  .024 .015 .014	0.571 .277 .144 .045 .053 .040 .026 .025 .022	 0.089 .014 021 038  052 049 041 037	0.623 .329 .207 .159 .103 .091 .076 .067	 0.015 062 091 104  108 093 080 079	0.689 .425 .291 .239 .188 .171 .148 .132 .119	 -0.100 156 170 184  185 156 132 117	0.759 .559 .432 .378 .321 .300 .263 .235 .217 .208	 -0.121 165 187 204  212 188 159 134	0.767 .590 .471 .418 .362 .338 .297 .267 .246 .236	0.342 .270 .207 .160 .129 .090 .064 .056	0.342 .294 .207 .194 .158 .110 .073 .056	0.230 .180 .128 .090 .063 .031 .011 .007	0.450 .375 .317 .266 .230 .177 .134 .107	0.078 .090 .049 .018 003 026 041 037 045	0.527 .451 .387 .334 .297 .238 .195 .168	 -0.188 114 095 084 081 093 101 088 097	0.630 .570 .503 .450 .409 .344 .290 .259 .244	0.232166148143133121126108115	0.660 .596 .534 .481 .442 .377 .323 .293
0.50	1 .967 .933 .900 .867 .800 .733 .667 .600 .533 .467 .400 .267 .133	.279 .179 .030 .004 .021 .047 .050 .054 .053 .052 .043 .037	.676 .313 .214 .065 .040 .008 021 039 043 045 042 041 035 029	050051068068104131088116115111100087086	.599 .287 .178 .136 .108 .073 .039 .014 .007 001 007 .001		.651 .400 .288 .226 .198 .151 .085 .070 .069 .058 .054 .053 .062		.682 .547 .425 .377 .343 .284 .234 .200 .183 .174 .162 .154 .146		.671 .579 .467 .418 .383 .321 .267 .236 .214 .205 .191 .181 .177 .168	306 .220 .172 .136 .020 009 036 051 061 057 044 033	.344 .276 .212 .164 .100 .042 001 050 053 050 035 027 023	-170 .126 .079 .052 009 046 069 091 107 102 086 071 065	.454 .370 .296 .244 .171 .105 .061 .009 .002 .004 .023 .025	071011001019080110128144149152144128112105	.533 .444 .367 .314 .167 .122 .089 .066 .061 .062 .074 .076		.650 .558 .490 .439 .357 .286 .233 .194 .167 .162 .161 .168		6533 .568 .508 .457 .374 .305 .253 .219 .195 .189 .187 .196
0.75	1 .978 .955 .933 .912 .889 .867 .845 .800 .756 .711 .667 .622 .578 .533 .446	 .063 .026 001 018  051 074 088 093 091 096 084 090	 .089 .055  009 034 046 064 070 081  082 082 077		 .164 .138  .052 .027 .013 007 015 028  032 030 034		.260 .209 .136 .106 .091 .065 .061 .040 .037 .037		.395 .344 .256 .227 .212 .189 .185 .165 .148 .153	- 260 - 274 - 278 - 281 - 290 - 299 - 305 - 309 - 310 - 304 289	.443 .391 .302 .273 .255 .230 .217 .205 .176 .185	221 .175 .115 .068 .029024051081086092089089089	 .203 .138 .090 .049 .015 012 066 070 077  076 076	076 .055 .010017057017017117142144148142138	.292 .223 .170 .120 .083 .056 .023 004 006 008 010 011	130114130129148152173197199200186182	.377 .297 .244 .190 .154 .126 .092 .063 .060 .059	286 - 266 - 256 276 - 279 280 - 280 - 271 - 252 - 231			

## TABLE VII .- CONTINUED

										β	= 5°										-11/2
										Right w	ing par	nel					1				
				. 1	Round—no	se air	foil								Sh	arp-nos	e airfo	11			
St	ation	α=	0.10	a=2	2.50	α=	5.20	α=8.6	50	α=10	0.10	α=0	.00	α=2	.7°	α=5	.20	α=8	.80	α=10	0.10
x/cr	y ₩/2	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl
0.25	933 .867 .800 .733 .667 .533 .400 .267	.144	0.697 .340 .186 .127 .074 .053 .026 .021 .017	0.164 .056 001 010 051 053 045 034	0.765 .421 .262 .201 .133 .128 .092 .079 .070	023 073 082  115 105 089	0.505 .350 .286 .224 .200 .166 .145 .129	 -0.058 117 154 166  190 171 144 120	0.869 .614 .463 .401 .333 .307 .263 .235 .212 .201	-0.079 138 164 184	0.879 .649 .501 .440 .374 .344 .298 .264 .241	0.398 .313 .241 .187 .150 .099 .065 .053	 0.413 .350 .291 .233 .193 .131 .084	0.271 .224 .164 .116 .084 .037 .010 .004	 0.518 .442 .373 .312 .266 .201 .148 .114	0.142 .121 .079 .042 .015 023 045 043	 0.608 .512 .439 .378 .333 .264 .205 .168	-0.102 062 058 061 073 096 109 109	0.709 .619 .543 .479 .436 .360 .296 .256	110 115 118	.634 .573 .502 .455 .382 .318
0.50	1 .967 .933 .900 .867 .800 .733 .667 .600 .533 .467 .400	228 .107 .050 .007017053072064053046	.697 .269 .149 .100 .069 .028 .010 .037 .050 .057 .060 .060 .053 .043	-108 .001 -036 -068 -117 -108 -126 -125 -119 -106 -091 -084	.032 .017 .009 .005 .002 .005	027 073 120 142 157 182 171 186 181 170 145 133 123			.796 .598 .469 .398 .361 .297 .238 .201 .178 .164 .155 .146 .140 .133 .128	162 205 220 228 242 269 236 266 267 252 228 180	.777 .634 .509 .439 .330 .270 .229 .204 .194 .186 .174 .166 .162	350 258 204 164 083 031 - 002 - 034 - 053 - 067 - 070 - 062 - 045 - 037	-1	- 215 .162 .115 .082 .014 - 030 - 062 089 106 118 121 067 076	522 .418 .349 .288 .207 .133 .080 .040 .010003008002 .017 .020	0-13 .023 .018 060 096 122 141 156 164 162 145 125	.598 .497 .409 .351 .266 .189 .138 .054 .049 .059 .058 .068	190 140 143 166 183 196 203 213 219 215 193 169 157		260 -206 -194 -192 -213 -224 -241 -243 -247 -247 -218 -193 -178	.702
0.75	.800 .756 .711 .667 .622	.094 .052 .021 .001 036 084 085 095 196 109 103 114	 .020 007 024 046 058 077  090 098 093		.166 		.318 .250 .175 .139 .117 .090 .078 .050 .050	228 - 239 - 246 - 247 - 258 - 279 - 284 - 284 - 284 - 282 - 275	.144 .134 .134 .134 .131 .132		.481 .412 .320 .286 .264 .234 .217 .199 .163 .165 .154	270 215 138 .095 .053005 -/037 074083094098096106	 .256 .188 .131 .085 .048 .016 018 .047 055 068 081 081	119 .097 .046 .014022101132141146154159	 .357 .284 .223 .167 .126 .092 .054 .010 001 015 023		433 .346 .285 .225 .191 .111 .076 .067 .055 043 .038 .026	214 196 209 231 248 254 262 263 262 256 243 253	 .571 .472 .407 .343 .269 .233 .192 .184 	286 264 254 249 286 292 302 302 300 225 283	.542 .470 .407 .348 .306 .271 .233 .192 .197 .186



										$\beta = 5^{\circ}$											
									Le	ft wing	panel										
				Roun	d-nose	airfoil									8	harp-no	se air	foil			
St	ation	α=-	-0.1°	<b>α=</b> 2	·5°	α=5	.20	α=8	.60	α=1	0.10	α=0	.00	α=2	.70	α=5	5.20	α=8	.80	α=]	10.10
x/c <sub>r</sub>	<u>y</u> w/2	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pı	Pu	Pl	Pu	Pı
0.25	1 •933 •867 •800 •733 •667 •533 •400 •267	0.161 .084 .050 .051  .036 .027 .026 .024	0.449 .209 .093 .061 .028 .028 .025 .021	 0.072 001 033 029  026 026 024 023	0.535 .294 .185 .156 .102 .097 .088 .079 .072		0.600 .392 .277 .236 .195 .186 .165 .148 .138	 -0.133 175 191 202  163 137 119 114	0.645 .505 .394 .352 .308 .288 .257 .234 .219	 -0.169 197 213 226  198 164 140 129	0.652 .548 .446 .405 .356 .335 .301 .275 .259	0.300 .237 .185 .147 .125 .096 .074 .073	 0.270 .237 .197 .163 .136 .100 .070 .055	0.175 .142 .101 .071 .053 .031 .014 .015	0.365 .307 .260 .222 .194 .148 .117 .099	0.045 .083 .046 .019 .003 016 027 023 039	 0.475 .414 .361 .318 .283 .231 .191 .169 .154	0 . 242 163 128 084 080 090 093 081 098	 0.542 .495 .455 .403 .370 .319 .277 .255 .238	 -0.300 218 191 161 103 101 088 109	0.604 .550 .498 .458 .426 .374 .331 .309 .288
0.50	1 .967 .933 .900 .867 .800 .733 .667 .600 .533 .467 .400 .267	131 .040 012 009 021 039 036 039 038 031 028 033	.442 .131 .050 .014 003 019 034 046 046 046 033 033 032		.517 .249 .158 .115 .095 .070 .041 .023 .017 .014 .011 .015 .012		•555 •356 •260 •219 •196 •157 •119 •099 •088 •085 •077 •076 •074 •070	228 252 241 252 256 226 224 192 175 159 148 149	.564 .481 .384 .349 .266 .223 .196 .188 .177 .177 .167 .154	255 - 275 - 275 267 - 278 - 285 - 267 - 264 - 240 - 299 - 189 - 171 - 171	537 .441 .393 .362 .308 .265 .240 .224 .218 .212 .206 .195 .187	- 246 .187 .144 .115 .053 .018 007 027 037 043 025 016	267 .208 .160 .120 .066 .017 -016 -039 -050 -050 -014 -026 -018 -019		.362 .289 .230 .187 .125 .073 .041 .014 003 005 .002 .020		467 .387 .333 .288 .216 .158 .118 .089 .071 .076 .085 .079	304 - 243 - 227 - 222 - 231 - 221 - 197 - 200 - 199 - 197 - 166 - 150 - 147	540 .469 .415 .372 .304 .247 .204 .180 .158 .156 .158	354 296 275 268 273 269 252 215 212 208 195 174 158 161	
0.75	1 .978 .955 .933 .912 .889 .867 .845 .800 .756 .711 .667 .622 .578 .533	019 009 030 045 085 092 093 076 082 071 083	017 004 059 077 089 095 089 085 080 074		.127 .090 .032 .010 015 018 031 	169 183 194 204 232 235 231 216 204 183 179	.118 .092 .081 .063 .062 .039 .053 .052 .061	- 284 - 298 - 299 - 299 - 306 - 314 - 317 - 314 - 303 - 294 - 280			.388 .357 .275 .251 .236 .217 .211 .205 .186 .101 .185	.174 .138 .071 .038 -005 	125 .078 .037 009 026 048 072 088 088 090 075 064 054	-019 .000 -038 -057 -089 -124 -137 -145 -144 -116	209 .148 .105 .061 .032 .011 015 037 030 025 015	165 143 150 141 156 179 196 192 199 185 177		324 304 286 287 297 297 295 260 230 222		- · 335 - · 335 330 - · 332 - · 308 - · 269 - · 250	 .433 .376 .330 .288 .262 .242 .224 .205 .221 .209  .194 .193 .181



TABLE VII. - CONTINUED

										β =	90										
					+0.17				Ri	ght win	g panel			Temp temb							
					Round-n	ose air	foil									Sharp-n	nose air	foil			1
Sta	ation	α=	-0.1°	α=	2.5°	α=	5.2°	α={	8.6°	α=	10.10	α=	0.0	α=2	2.7°	α=	5.2°	α=	8.8°	α	=10.1
x/cr	y w/2	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl
0.25	.933 .867 .800 .733 .667 .533 .400 .267	0.273 .162 .100 .072  001 018 017	0.770 373 214 .147 .085 .055 .014 .004 003 003	0.215 .094 .025 .015 	0.879 .492 .317 .244 .166 .144 .104 .082 .069			-0.010 076 127 138 178 190 162 129	0.976 .687 .507 .433 .355 .323 .270 .234 .207 .188			0.450 .355 .278 .213 .170 .106 .062 .043	0.471 .402 .336 .271 .220 .146 .090 .054	0.318 .258 .194 .142 .103 .046 .009 005	0.595 .497 .419 .348 .297 .218 .154 .112			 -0.052 015 024 027 056 091 118 111 119	0.768 .676 .583 .510 .455 .371 .295 .243		
0.50	1 .967 .933 .900 .867 .800 .733 .667 .600 .533 .467 .400 .267 .133	.238 .102 .054 .008 .016 .062 .039 .084 .095 .095 .087 .071	.762 .290 .159 .113 .078 .033 007 040 056 062 076 080 067 060	- 158 - 038 - 018 - 053 - 078 - 110 - 117 - 135 - 141 - 139 - 120 - 104 - 092	.846 .417 .279 .215 .177 .126 .078 .042 .023 .007 002 .002 .002				.896 .646 .499 .423 .383 .313 .249 .206 .157 .148 .139 .126 .126				.464 .378 .292 .232 .155 .083 .029 011 041 056 066 059 039	 .245 .189 .133 .099 .027 019 053 083 105 129 129 120 085	.588 .477 .382 .317 .231 .150 .093 .051 .003 008 007 .016			125 - 088 - 095 - 106 - 142 - 168 - 191 - 206 - 223 - 237 - 245 - 230 - 208 - 195	.784 .671 .569 .502 .400 .309 .241 .183 .123 .111 .108 .112		
0.75	1 •978 •955 •933 •912 •889 •867 •845 •800 •756 •711 •667 •622 •578 •533	- 103 .054 .021 - 001 - 070 - 089 - 103 - 106 - 115 - 111 - 128	.152 .113 .027 003 020 046 057 070		.267 .194  .119 .088 .066 .044 .025 003			- 198 - 206 - 217 - 222 - 240 - 255 - 256 - 272 - 274 - 276	.473 .408 .305 .266 .243 .210 .194 .167			.291 .236 .155 .111 .067 .006 029 	-302 -217 -156 -108 -067 -033 -001 -032 -042 -056 	-144 -118 -057 -023 -014 066 -095 126 -135 -143 -152				-127 -121 -146 188 -218 -234 -255 -260 -266 -270			



## TABLE VII. - CONCLUDED

								-	Le	ft wing	panel										
				I	Round-no	ose airí	?oil								S	harp-no	se airf	oil			
Stat	cion	α=	-0.1°	α=2	2.50	α=5	5.20	α=8	3.60	α=10	0.10	α=(	0.00	α=2	2.70	α=	5.20	α=8	3.80	α=1	0.10
x/cr	<u>y</u>	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pı	Pu	Pl -	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl	Pu	Pl
0.25	1 .933 .867 .800 .733 .667 .533 .400 .267	0.140 .080 .058 .062  .049 .041 .040	0.375 .168 .077 .056 .033 .012 .037 .033 .032	 0.051 009 027 017  011 013 011 017	0.465 .260 .120 .140 .104 .102 .095 .086 .079			 -0.173 201 207 201  140 117 104 107	0.550 .460 .380 .345 .305 .289 .262 .243 .232			0.269 .219 .174 .146 .126 .102 .083 .085	0.228 .210 .178 .151 .128 .096 .072 .061	0.128 .107 .076 .051 .038 .019 .007 .009 022	0.294 .256 .216 .188 .160 .124 .099 .085			 -0.310 210 147 058 071 072 074 062 092	0.509 .465 .422 .388 .362 .318 .286 .268		
0.50	1 .967 .933 .900 .867 .800 .733 .667 .600 .533 .467 .400 .267 .133	 .103 .037  .016 001 007 019 017 016 017 013 012 018	.355 .086 .016 005 011 025 031 030 027 025 018 022 028	 - 011 - 080 - 085 - 086 - 084 - 089 - 088 - 079 - 074 - 072 - 064 - 062 - 068	.437 .207 .132 .101 .086 .074 .041 .025 .025 .025 .020 .023 .021 .018				.464 .444 .361 .323 .295 .259 .202 .193 .183 .183 .180 .164 .162			 .216 .154 .126 .104 .050 .022 0 011 019 021 014 008 004 008	 .199 .172 .126 .094 .019 .010 015 029 035 035 037 015 010 015	042 .014 .004011049064085088092096070078					476 .415 .377 .342 .284 .238 .208 .177 .174 .176 .189 .181		
0.75	1 .978 .955 .933 .912 .889 .867 .845 .800 .756 .711 .667 .662 .578 .533		032 046 081 - 093 - 095 - 096 - 088 - 087 079 - 079 - 0766 - 067	101 - 126 - 139 - 146 156 - 160 - 158 - 151 - 133 - 134 - 112 - 116	091 .067 008 005 015 024 023 031 023 020 015							140 .105 .046 .018 - 011 050064075077055	 .078 .033 0 041 053 069 087 091 091 086 		 -119 .078 .042 .005 018 053 066 051 038 038			369 - 347 - 325 319 - 326 - 321 287 - 241 - 232 - 225 - 210			



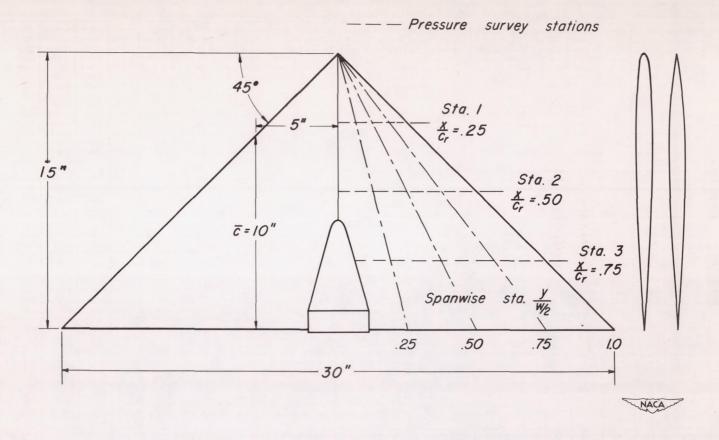
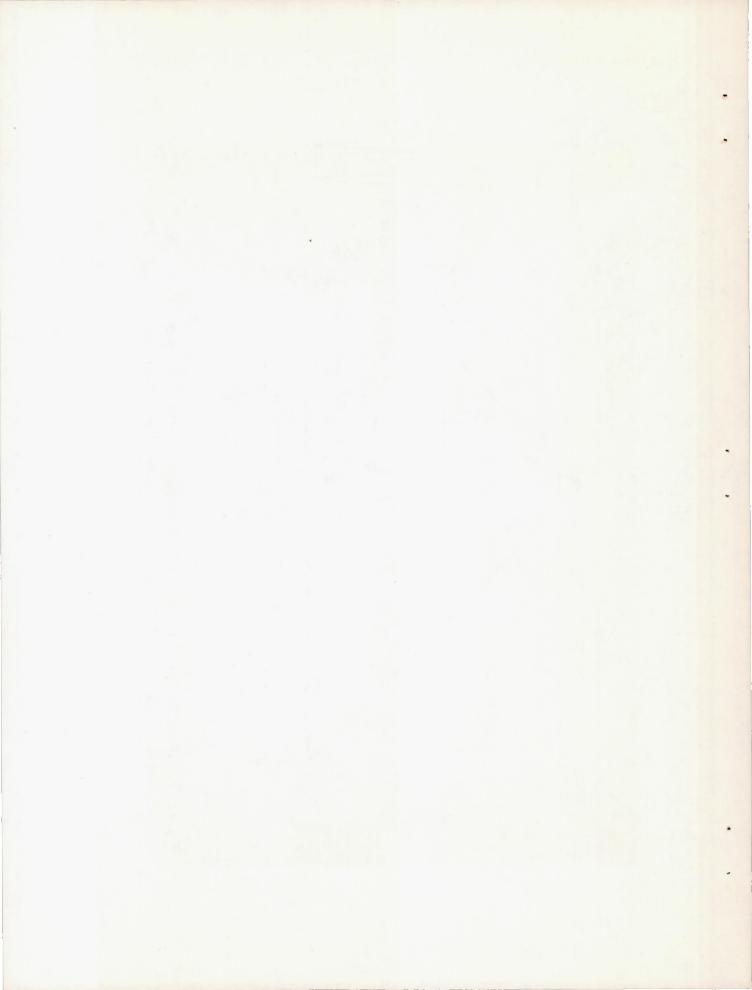


Figure 1. — Dimensional sketch of triangular wing showing both the round-nose airfoil section (NACA 0006-63) and the sharp-nose airfoil section.



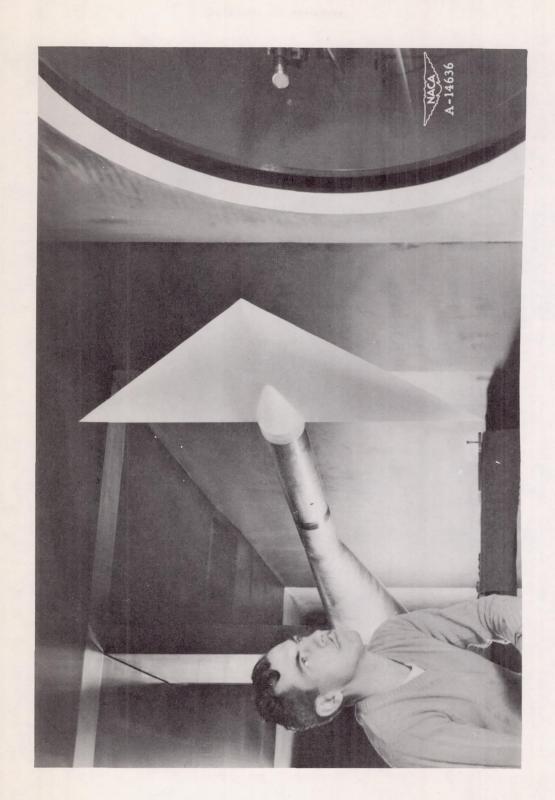
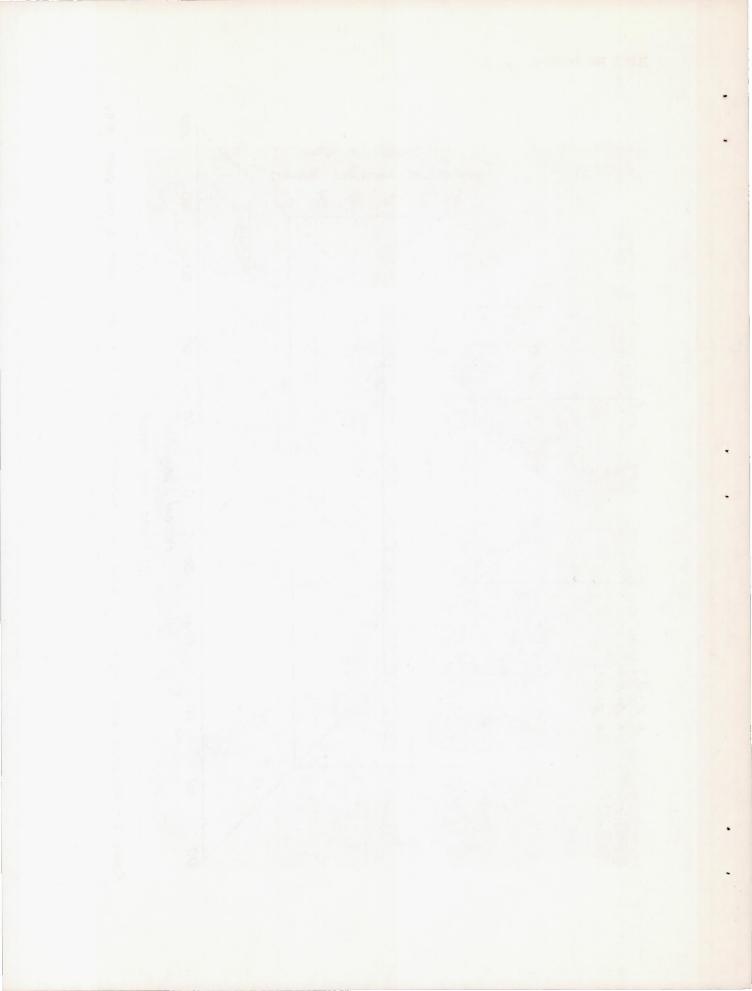


Figure 2.- Model mounted in the Ames 6- by 6-foot supersonic wind tunnel.



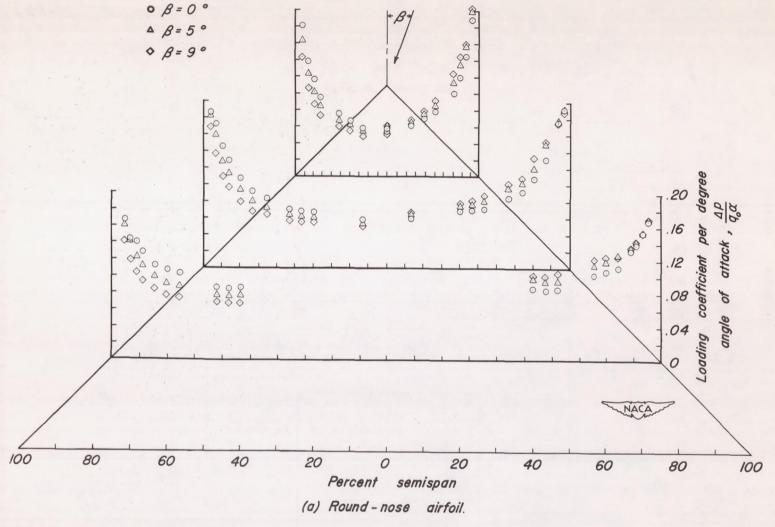


Figure 3. – Variation with angle of sideslip of experimental load distribution over a triangular wing at  $2.5^{\circ}$  angle of attack. M = 1.20.

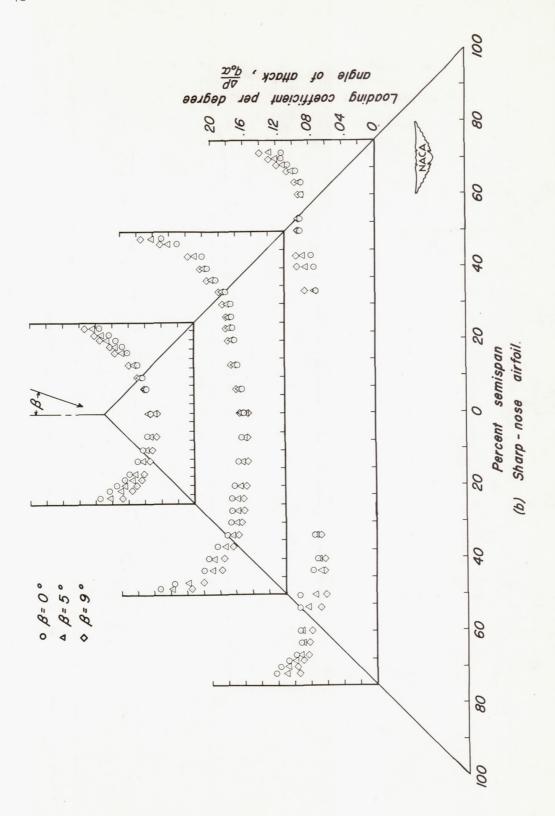
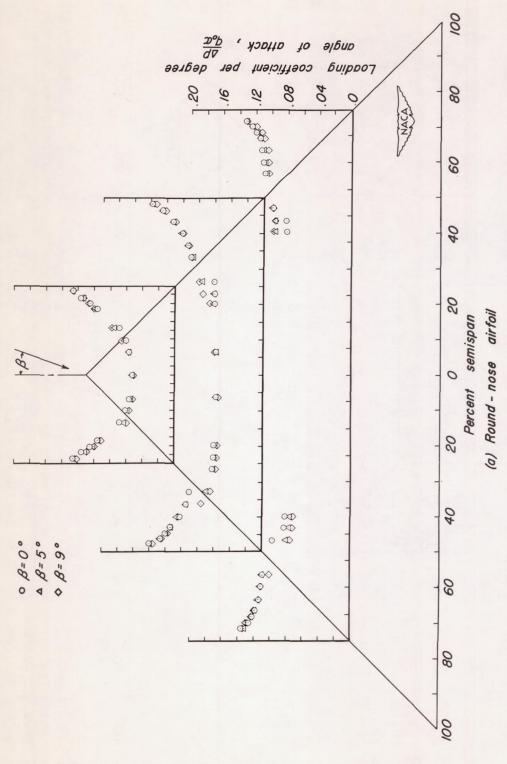


Figure 3. - Concluded.

7



a triangular wing OVER of sideslip of experimental load distribution at 8.6° angle of attack. M=1.20. angle Figure 4. - Variation with

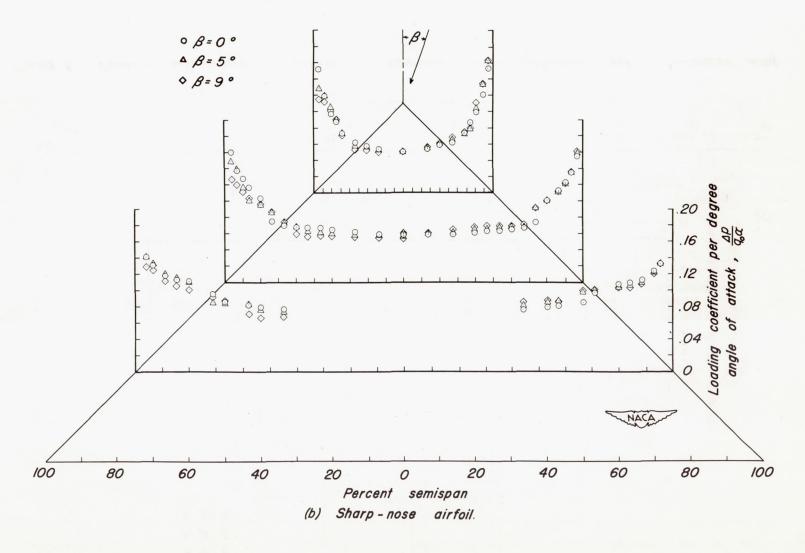


Figure 4.- Concluded.

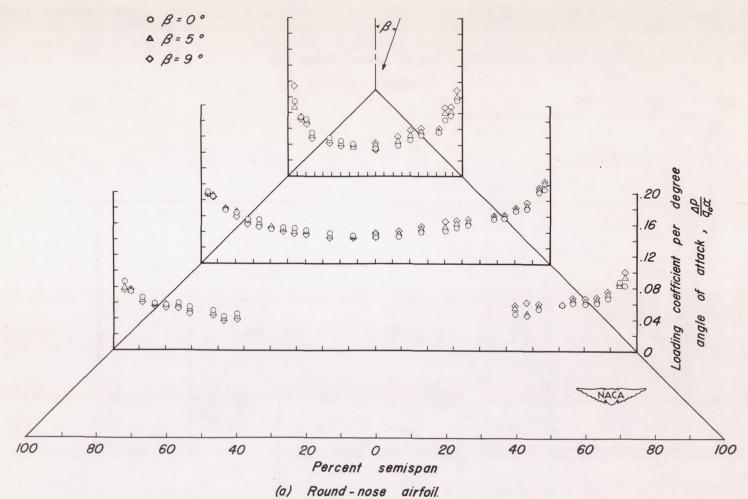


Figure 5. — Variation with angle of sideslip of experimental load distribution over a triangular wing at 2.5° angle of attack. M=1.70.

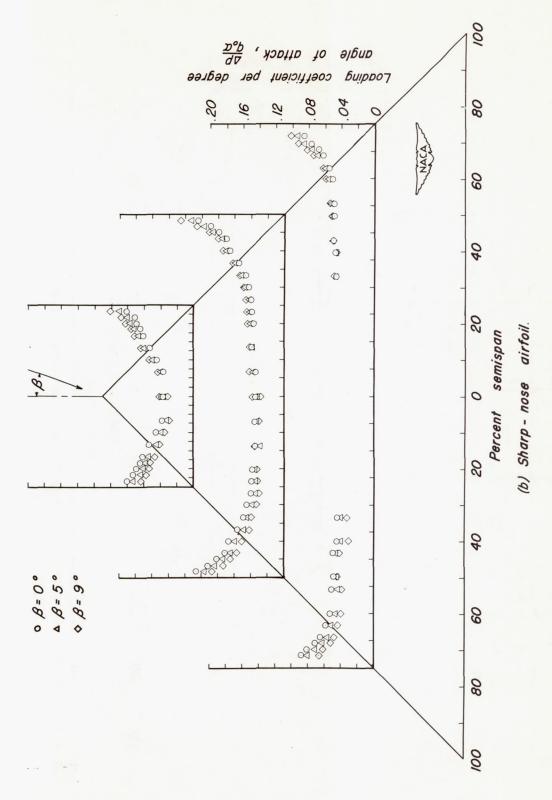


Figure 5.- Concluded.

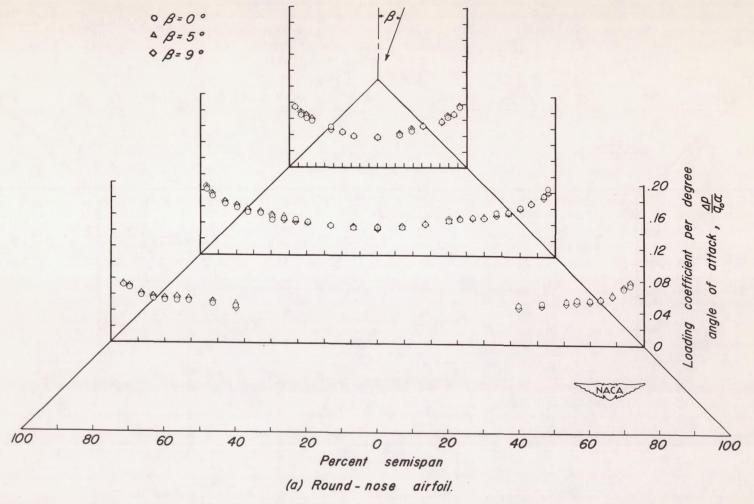


Figure 6. — Variation with angle of sideslip of experimental load distribution over a triangular wing at  $8.6^{\circ}$  angle of attack. M = 1.70.

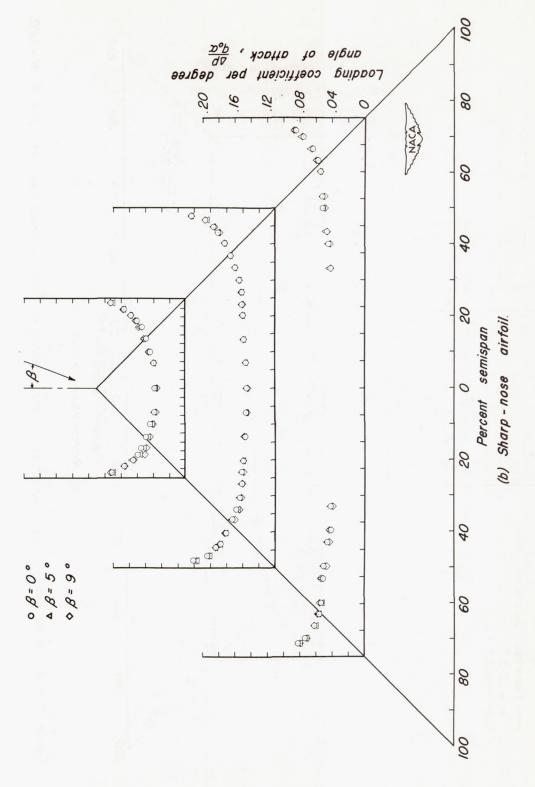


Figure 6. - Concluded.

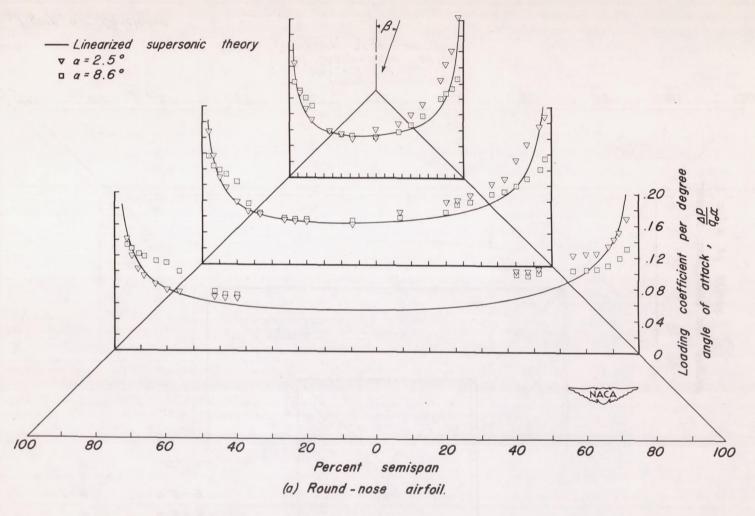


Figure 7. — A comparison of the experimental and theoretical load distributions at  $\beta = 9^\circ$ . M = 1.20.

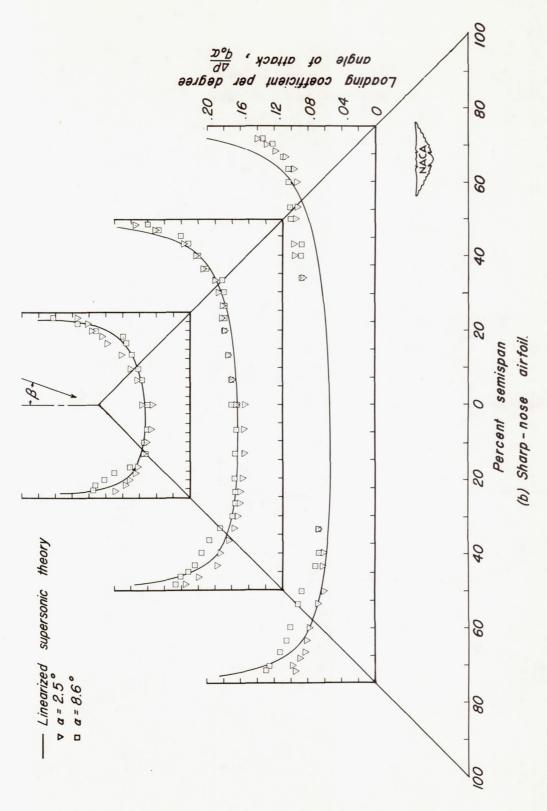


Figure 7.- Concluded

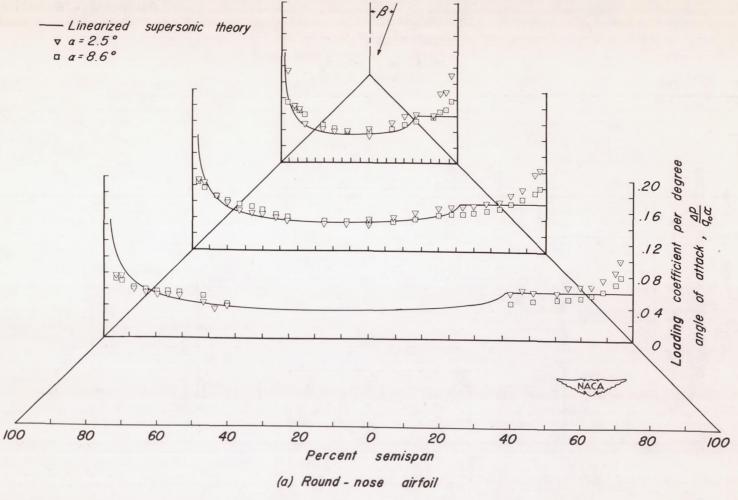


Figure 8. — A comparison of the experimental and theoretical load distributions at  $\beta = 9^\circ$ . M = 1.70.

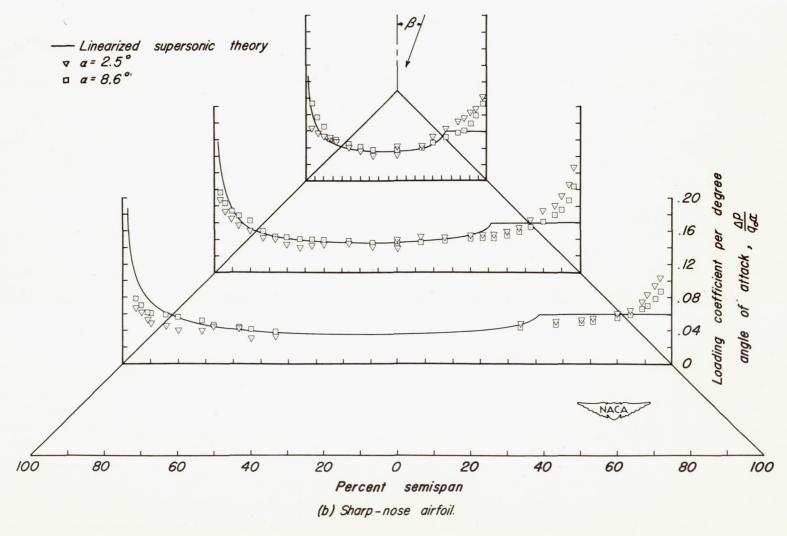


Figure 8.- Concluded.

NACA - Langley Field, Va.